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J. L. Wolf

SEASAT STUDY DOCUMENTATION

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1.0 INTRODUCTION

This document contains the results of the Boeing Company study effort in response to Article Ia of the Statement of Work of Jet Propulsion Laboratory Contract No. 953801.

2.0 SUMMARY

The proposed spacecraft design, developed for the SEASAT-A, is based upon the technical and operational requirements of the sensor equipment supplied by JPL during the course of the study supplemented by guidance received from JPL during the technical review meetings. The proposed spacecraft reflects the Boeing-built flight proven STP P72-1 Spacecraft design.

The proposed Spacecraft consists of a Bus Module, containing all subsystems required for support of the sensors and a Payload Module containing all of the sensor equipment. The two modules are bolted together to form the spacecraft. Electrical interfaces are accomplished via mated connectors at the interface plane. This approach was chosen because it permits independent parallel assembly and test operations on each module up until mating for final spacecraft integration and test operations.

A high probability of mission success is predicted based upon the proposed use of redundant subsystems to minimize single point failures and the use of mature flight proven or fully qualified subsystem equipment. All subsystem concepts are flight proven. This approach was used on the highly successful STP P72-1 spacecraft, which had a 1-year life requirement. The spacecraft is currently fully operational and meeting all performance requirements after 18 months in orbit.

The Atlas F/Burner II was selected as the candidate launch vehicle for the proposed SEASAT spacecraft. The Atlas F/Burner II is a flight proven Air Force launch vehicle, has adequate performance margin, is believed to be comparable or lower in cost than a Delta, and requires no spacecraft/launch vehicle interface development for use in the SEASAT program. However, should the large imaging radar become a firm payload, the Delta would be the recommended launch vehicle because of the larger diameter of the Delta heat shield as compared to the Burner II heat shield. Spacecraft configurations compatible with each launch vehicle were developed.

Proposed program schedules recognize the need to refine sensor/spacecraft interfaces prior to proceeding with procurement, reflect the lead times estimated by suppliers for delivery of equipment, reflect a comprehensive test program, and provide flexibility for unanticipated problems. This approach provides high confidence in achieving proposed schedules.

Program plans were developed to identify the approach to functions required to assure the technical success of the proposed program within the costs and schedule. These plans reflect the experience of the Contractor on JPL (MVM) and Air Force (STP P72-1, S3 and STP 70-1) spacecraft programs. They recognize and address the many organizational, management, design, test, schedule and operational interfaces which exist on a spacecraft program.

3.0 MISSION AND SPACECRAFT SYSTEM DESCRIPTION

3.1 CONFIGURATION

The proposed spacecraft configuration is shown in Figure 3.1-1 and consists of the "Bus Module" and "Payload Module". All experiments and related electronics are installed on the "Payload Module" and all housekeeping systems such as power, attitude control and TT&C are installed on the "Bus Module". The mechanical interface between the Bus and Payload modules is accomplished via a bolted joint. The spacecraft is separated from the Burner iI (STP P72-1) using three (3) pyrotechnic bolts and three (3) springs. This separation system is identical to that flown on Burner II launches and which was used to successfully separate the P72-1 satellite in 1972. Power, data, and control signal interfaces between the Bus and Payload modules are accomplished via mating electrical connectors at the interface plane.

The "On Orbit" configuration with stick array atnennas and solar panels deployed is shown in Figure 3.1-2. This configuration satisfies all look angle requirements of the sensors (with clear fields of view).

This spacecraft configuration satisfies all mission requirements and is within the performance capability of the launch vehicle.

An alternate configuration has been developed which uses the Delta launch vehicle - see Figure 3.1-3. This spacecraft configuration is similar to the proposed configuration except that the Bus Module structure is cylindrical in shape rather than hexagonal. This change is required because of the interface with the Delta 3731A attachment fitting. The interface between the Payload and Bus Modules is a bolted joint similar to the proposed configuration. Separation of the spacecraft from the Delta attachment fitting is accomplished by the use of an existing design Marman clamp system.

The "On Orbit" configuration with stick array antennas and solar panels deployed is shown in Figure 3.1-4.

This alternate configuration also satisfies all mission requirements and is within the performance capability of the Delta launch vehicle.

A system functional block diagram and a system physical block diagram, applicable to spacecraft designed for either launch vehicle, are shown in Figures 3.1-5 and 3.1-6, respectively.

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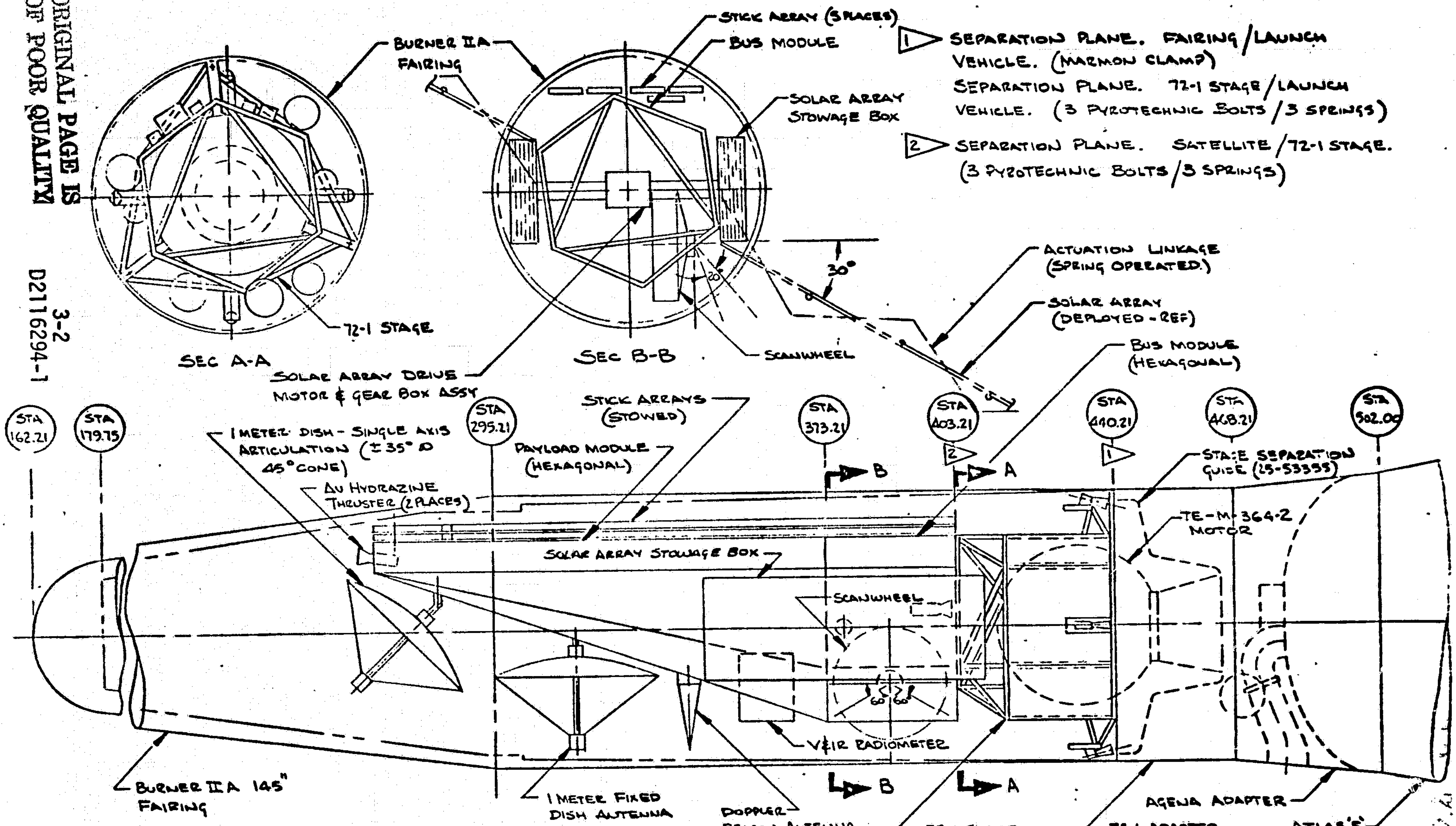

- SEASAT BASELINE
CONFIGURATION - ATLAS 'F' / BURNER II -

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FIGURE 3.1-1

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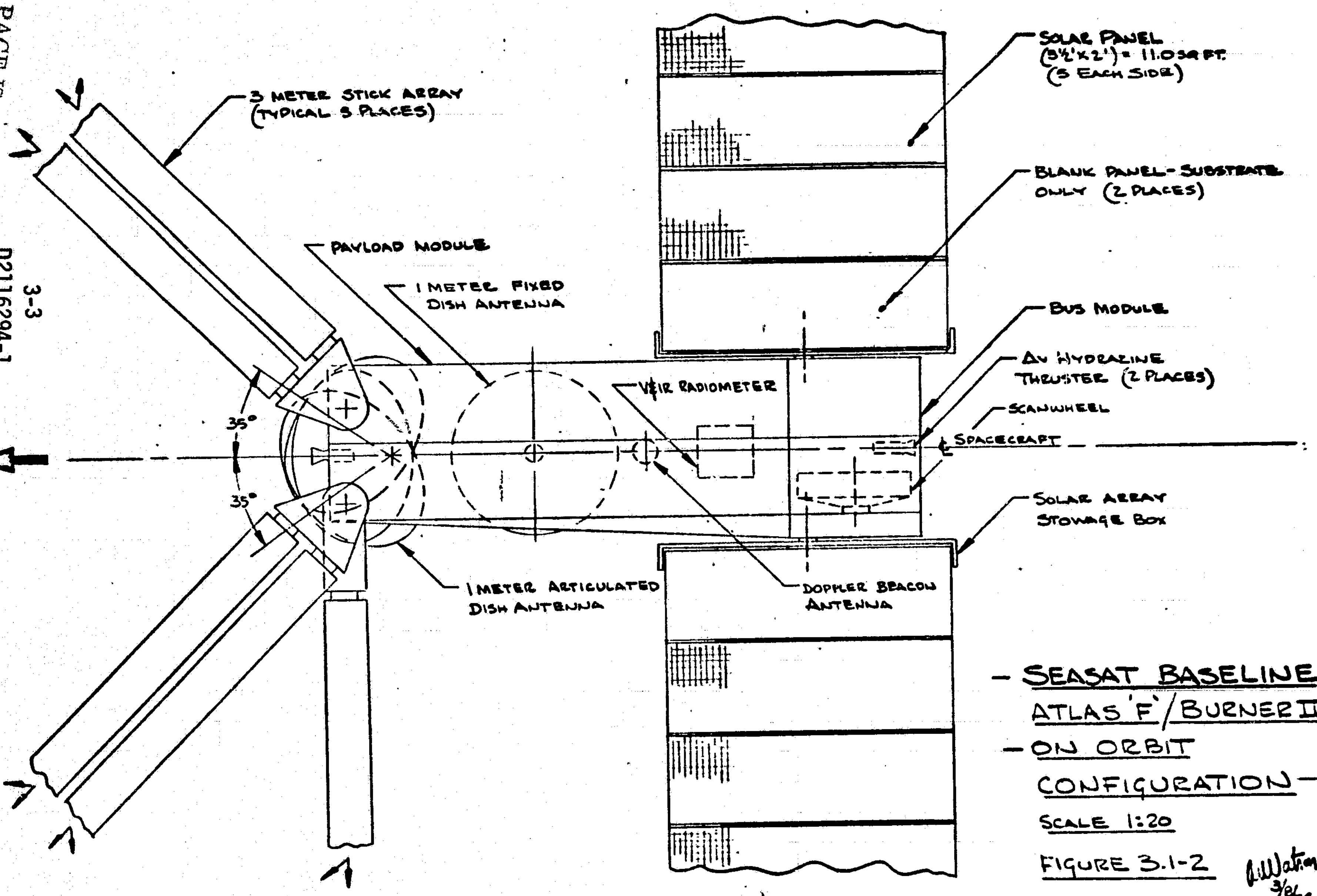

- SEASAT BASELINE -
ATLAS 'F' / BURNER II
- ON ORBIT
CONFIGURATION -
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FIGURE 3.1-2

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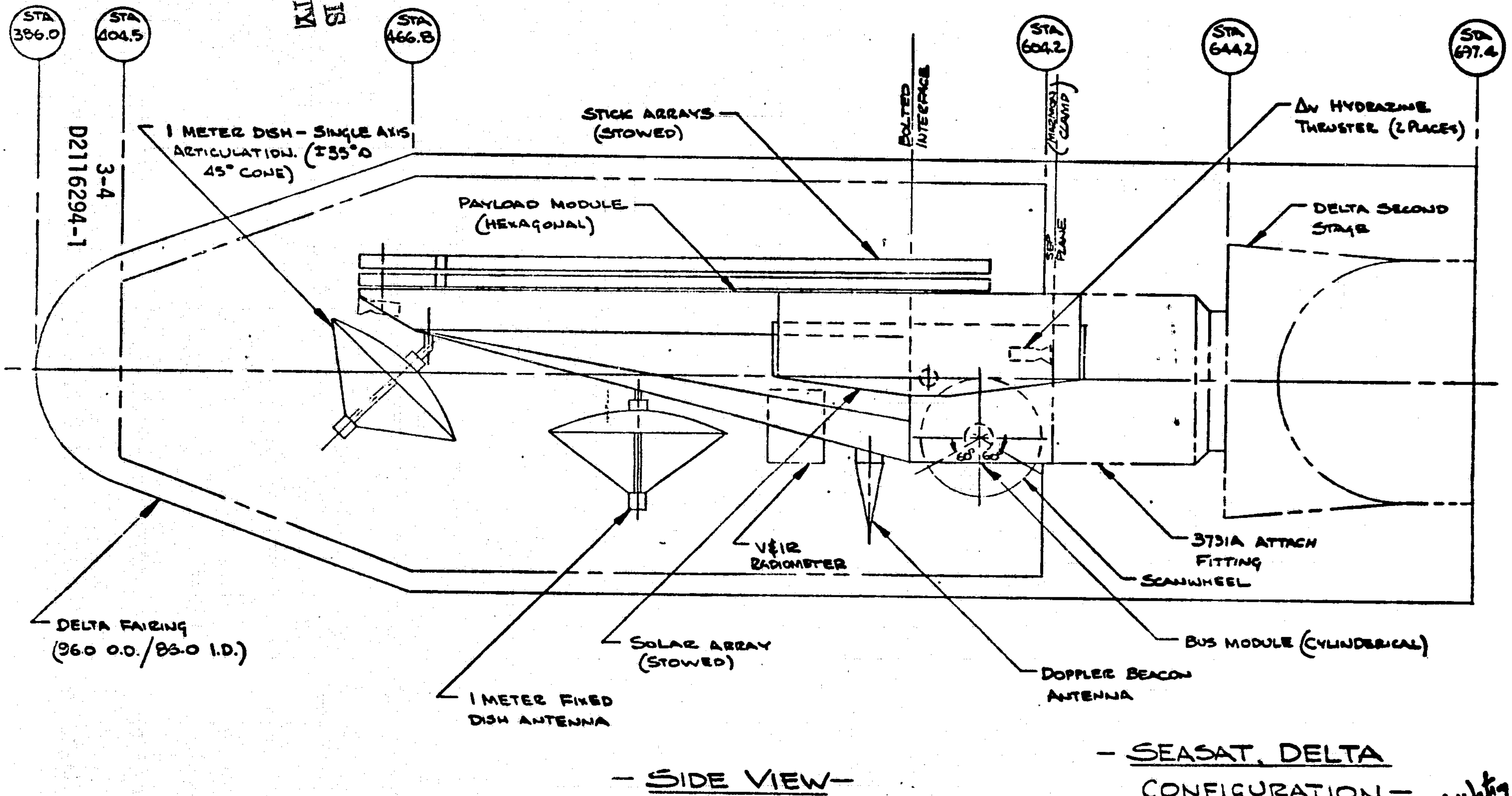

- SEASAT, DELTA
CONFIGURATION -
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FIGURE 3.1-3 SH 1 OF 2

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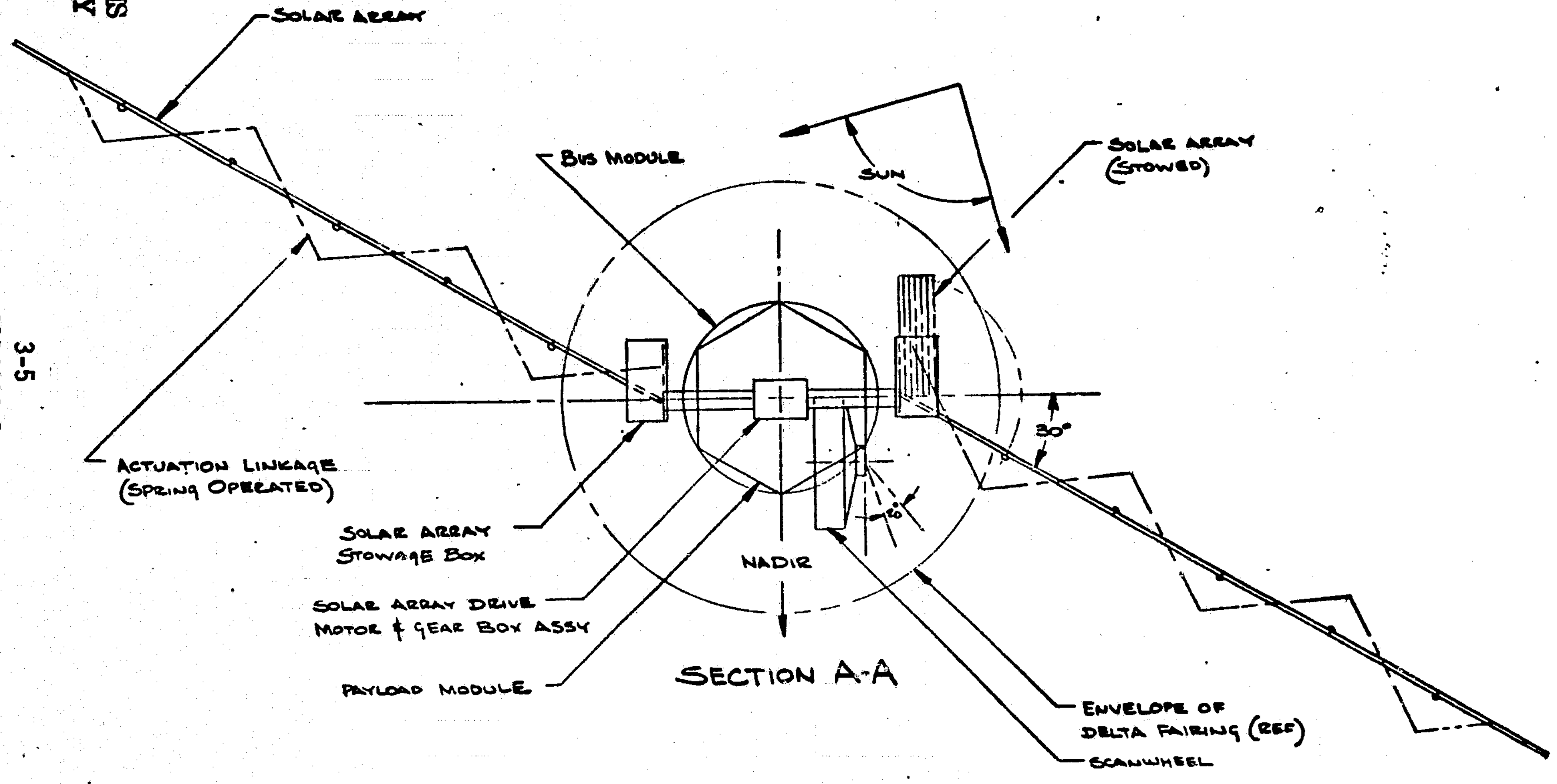

SEASAT, DELTA CONFIGURATION — SCALE 1:20

FIGURE 3.1-3 SH 2 OF 2

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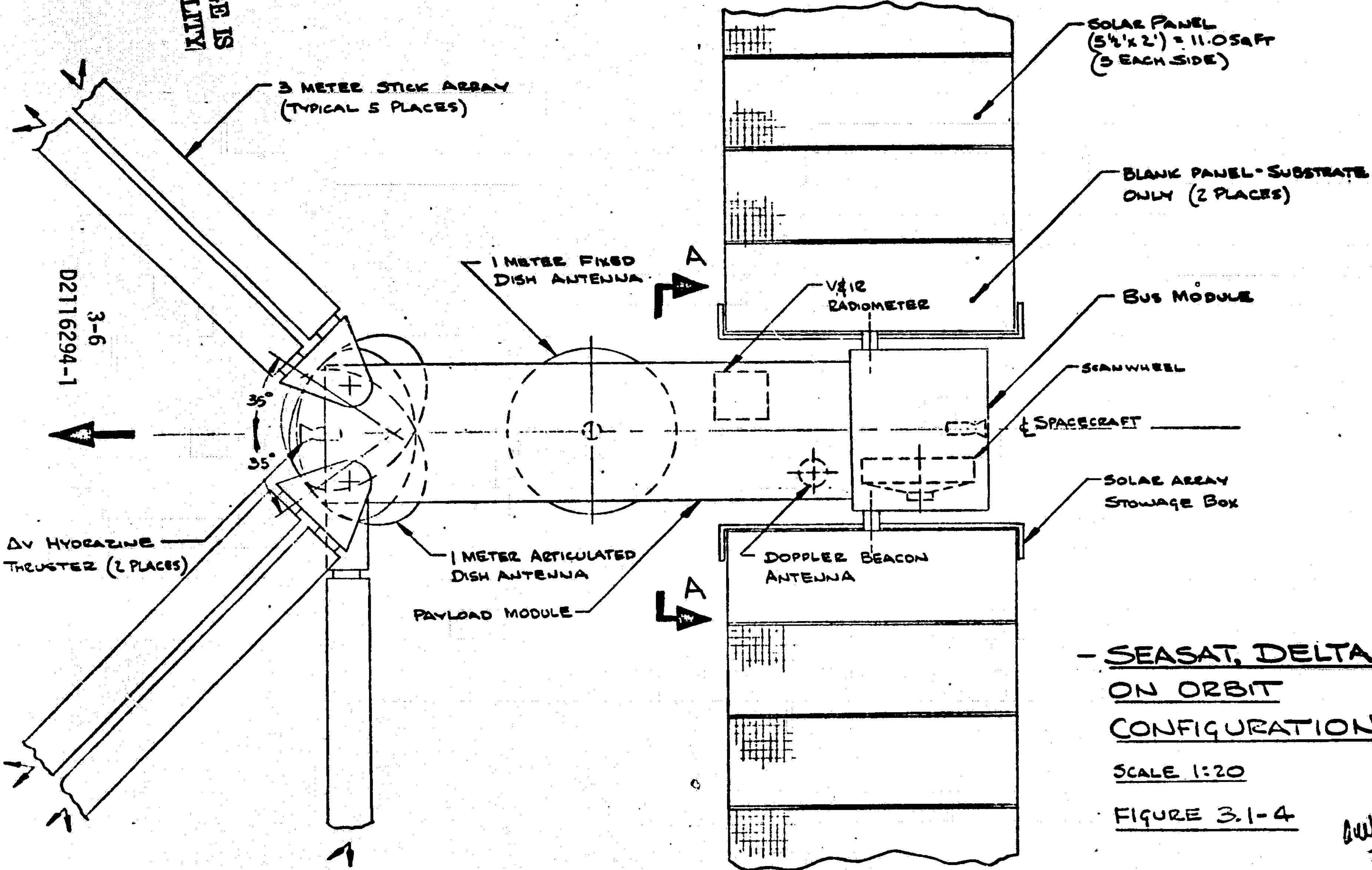

- SEASAT, DELTA
ON ORBIT
CONFIGURATION -

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FIGURE 3.1-4

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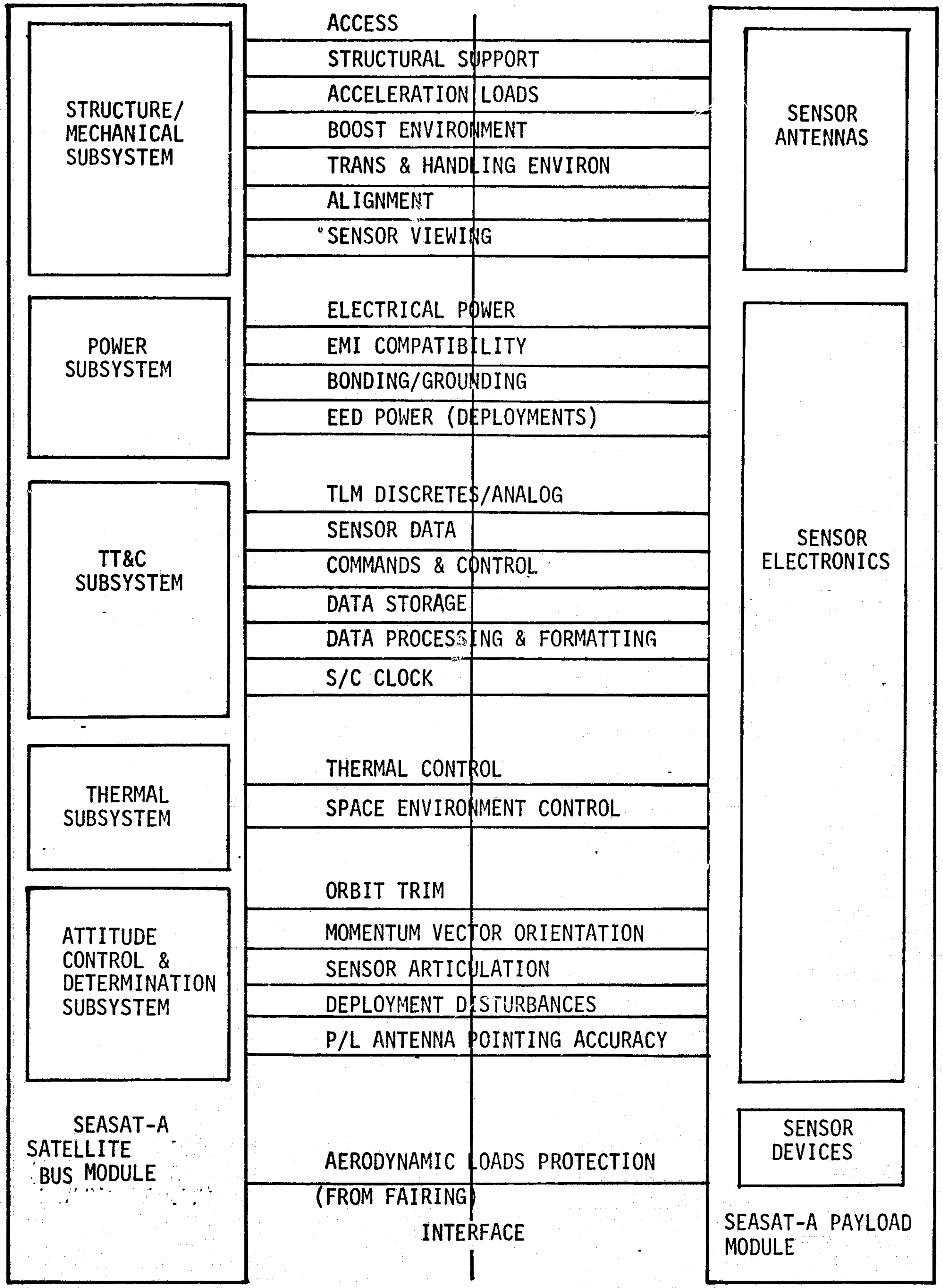

FIGURE 3.1-5 System Functional Block Diagram

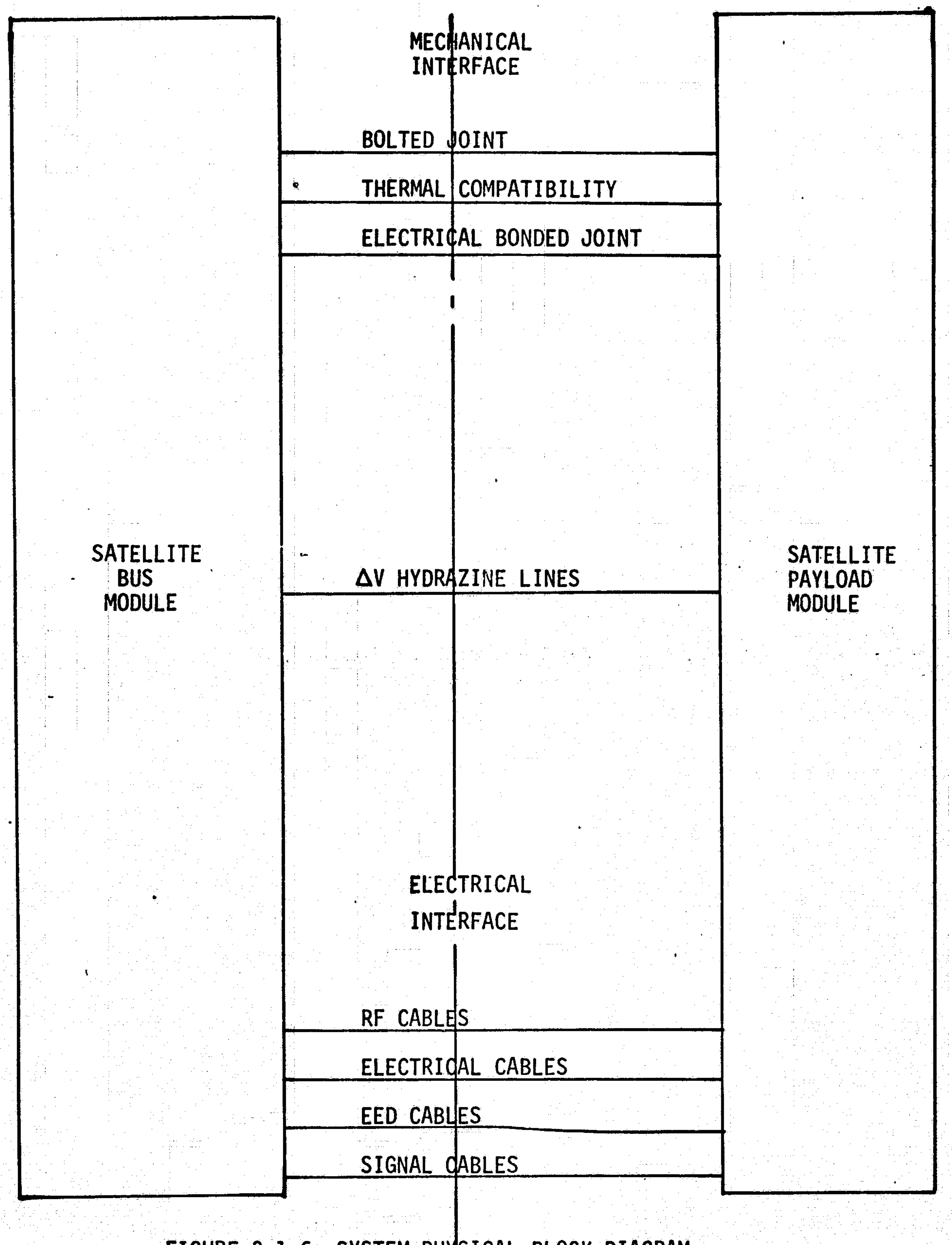

FIGURE 3.1-6: SYSTEM PHYSICAL BLOCK DIAGRAM

3.2 MASS PROPERTIES

A weight summary of the proposed SEASAT configuration is shown in Table 3.2-1. This weight data was derived from:

- Evaluation of vendor proposals.
- Weight calculations from preliminary design layouts.
- Actual weights of off-the-shelf components.
- Estimates based on similar systems or installations.

As shown in the table, the satellite weight vs. the Atlas 'F/Burner II capability is 1442 lbs. vs. 1800 lbs. This provides a large allowance for growth and ballast.

TABLE 3.2-1: WEIGHT STATEMENT

o SENSORS (LESS ANTENNAE)	351 LBS.
o SPACECRAFT STRUCTURE (BUS MODULE/PAYLOAD MODULE)	200 LBS.
o POWER SUBSYSTEM (ARRAY, BATTERIES, DRIVE, ETC.)	336 LBS.
o MOMENTUM BIAS ATTITUDE CONTROL SUBSYSTEM	100 LBS.
o THERMAL CONTROL	40 LBS.
o ANTENNA SUBSYSTEM (INCLUDING DRIVES)	195 LBS.
o TELEMETRY, TRACKING AND COMMAND (TT&C) SUBSYSTEM	90 LBS.
o ELECTRICAL WIRING/CONNECTORS	90 LBS.
o HYDRAZINE SUBSYSTEM	40 LBS.
SPACECRAFT TOTAL	1442 LBS.
ATLAS 'F'/BURNER II LAUNCH CAPABILITY FOR SEASAT MISSION TO 1000 KM AT 110^{o} INCLINATION	1800 LBS. (APPROX.)

3.3 ENVIRONMENTAL REQUIREMENTS AND STRUCTURAL DESIGN CRITERIA

3.3.1 ENVIRONMENTAL REQUIREMENTS

Environmental requirements consistent with an Atlas F/Burner II launch were used in the design and analysis of the proposed spacecraft system. These environments are essentially identical to the design values used for all previous Burner II and Burner II related STP missions. They present a very low technical risk and permit a cost effective system design through the use of many off-the-shelf flight proven equipment components. Significant environments considered in structural design and equipment procurement include boost acceleration, acoustic and resulting random vibration, and nose fairing internal temperatures.

The maximum expected quasi-steady accelerations during Atlas F boost are shown in Table 3.3-1. The maximum longitudinal acceleration during Burner II motor burn is 5.0 g's and occurs with negligible lateral accelerations.

TABLE 3.3-1: MAXIMUM ACCELERATIONS DURING ATLAS F BOOST

CONDITION	FLIGHT EVENT	LONGITUDINAL	LATERAL
Maximum Lateral	Maximum Wind Excitation (Max q)	2.5	± 2.5g
Maximum Longitudinal	Burnout of Booster and Sustainer Engines	8.0g	± 0.7g
Minimum Longitudinal	Launch, BECO, SECO	-2.0g	± 0.7g

Predicted acoustic environments external and internal to the nose fairing are presented in Figure 3.3-1. External levels are based on Report No. GDC-ANR67-005, "Atlas E/F Boosters and BMRS-A Criteria for Payload Designers," 12 June 1970 (Revision B). The internal levels were established by reducing this external environment to reflect the noise reduction measured during Burner II nose fairing acoustic tests.

Envelopes of maximum expected random vibration levels for equipment and payload components are presented in Figure 3.3-2. The qualification test status of all off-the-shelf flight proven components will be evaluated for acceptability and additional testing specified where required. All new or extensively modified components will be vibration tested to a qualification test level 3 dB higher than shown in Figure 3.3-2. Qualification test duration will be three minutes.

The SESP P72-1 nose fairing provides a high level of thermal protection during boost and eliminates any requirement for additional protection during this period. Maximum fairing internal surface temperatures vs. time are presented in Figure 3.3-3 and spacecraft component and structural temperature rise under this environment are not significant.

Shock levels generated by the flight proven nose fairing and Burner II stage separation devices do not require qualification at the component level.

Components qualified for the vibration and acceleration levels are adequately qualified for this low risk environment. Pyro shock testing by ordnance firing at the spacecraft level during acceptance testing will provide assurance that these shock levels do not degrade payload or equipment performance.

3.3.2 STRUCTURAL CRITERIA AND REQUIREMENTS

Structural criteria and requirements are based on experience gained in the development of cost effective, reliable designs for previous Atlas F/Burner II missions. The criteria emphasizes the use of conventional flight proven structural concepts that are readily analyzed and can be verified with a minimal structural test program. Structural sizing is based primarily on stiffness requirements and permits the use of high safety factors for essentially no structural weight increase. The high safety factors and conservative design permit the structure to be confidently developed without a costly static test program or a structural development model.

Primary criteria affecting the structural design of the bus and payload modules are:

- Stiffness compatible with meeting a total stack minimum cantilever frequency of 7.5 Hz above Atlas Station 502.
- Bending moments, axial loads, and shears based on the following boost acceleration conditions:

Liftoff	-2 g axial, ± 0.7 g lateral
Max. Q	+2.5g axial, ± 2.5 g lateral
Max. Boost Axial	+8.0g axial, ± .70g lateral

- Maximum Aerodynamic Bending Moment = 930,000 in.lbs. at the base of the nose fairing (Atlas Station 440)
- Factors of Safety for all New Structure

Yield Safety Factor	1.50
Ultimate Safety Factor	2.00

- Factors of Safety for Existing Structurally Qualified Designs

Yield Safety Factor	1.10
Ultimate Safety Factor	1.25

- Nose Fairing Separation Altitude = 330,000 feet minimum and acceleration at the time of separation = 2.0g maximum.
- Nonmagnetic materials for all structures.
- Outgassing controlled by material selections based on JPL Specification ZPP-2062-MPL-A, "Preferred Materials and Processes List for Spacecraft Applications."

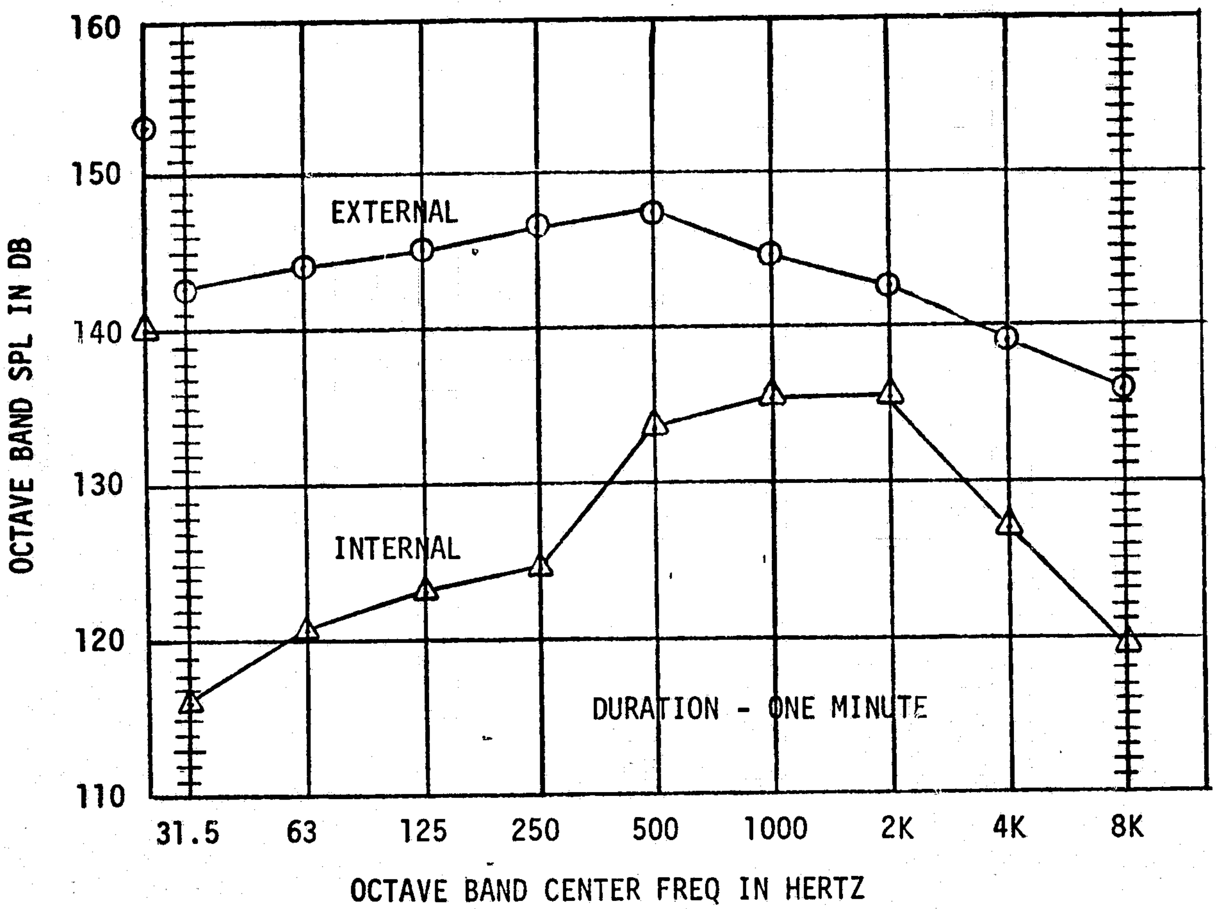

FIGURE 3.3-1: MAXIMUM EXPECTED ACOUSTIC LEVELS

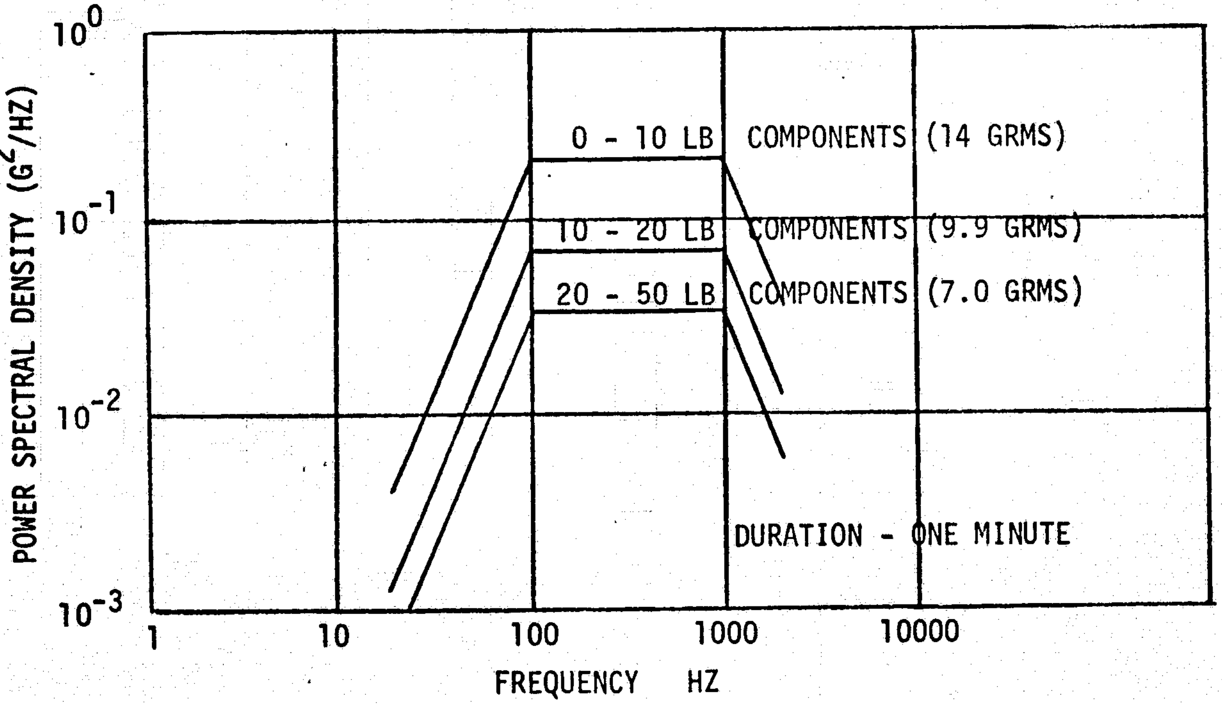

FIGURE 3.3-2: MAXIMUM EXPECTED RANDOM VIBRATION LEVELS

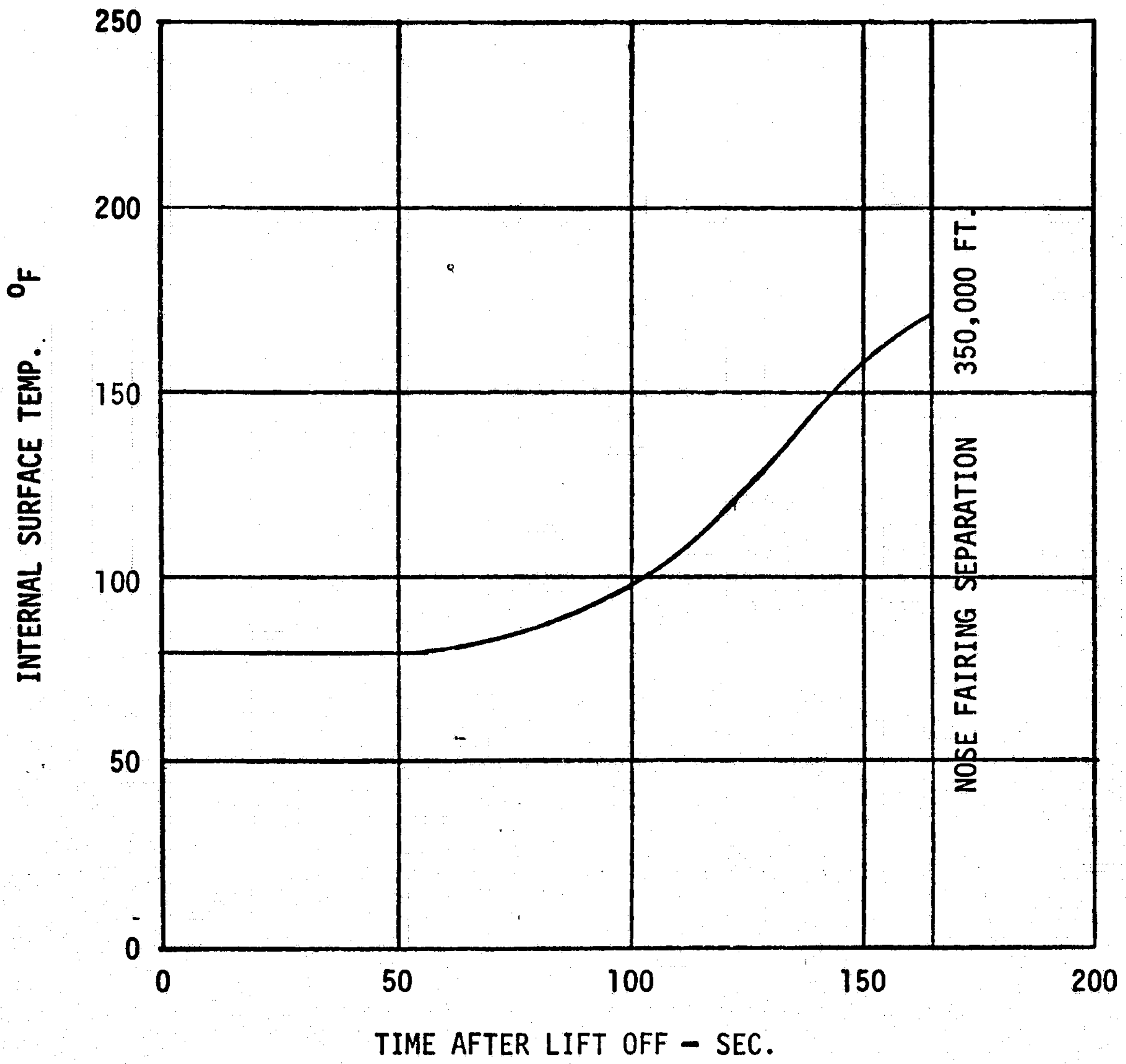

FIGURE 3.3-3: NOSE FAIRING MAXIMUM INTERNAL SURFACE TEMPERATURE

3.4 PERFORMANCE

Candidate launch vehicles for SEASATS include Delta and Atlas F/Burner II with the latter being the baseline for this study.

This section presents the launch vehicle performance characteristics and a summary of orbit design considerations.

3.4.1 Launch Vehicle Performance

Preliminary requirements for the launch system are to carry a throw weight of approximately 2000 lb to a highly inclined orbit with altitude of about 1000 km.

Performance for the Atlas F/Burner II and for the Delta is shown in Figure 3.4-1 for the polar orbits with altitudes in the range of interest.

The Atlas F nose fairing is jettisoned at 350,000 ft and the nose fairing adapter and Burner II adapter are jettisoned with the sustainer. Payload weights shown include the payload adapter.

The Delta is a straight eight model with solid motor thrust augmentation. The number of Castor II solid motors is given by the second digit of the model identification label on each curve (i.e, 3, 6 or 9 Castor II strapon motors).

For non-polar orbits, Figure 3.4-2 gives the effect on payload capability of changing orbit inclination. Orbit inclination that are greater than 80^o can be easily obtained within the range safety requirements of the WTR. If desired, Atlas launches from the SLC-3 complex can achieve lower inclinations and, thus, greater payloads.

3.4.2 Orbit Characteristics

Desired orbital characteristics include the following:

1. Wide seasonal variation (achievable with inclinations of 108^o to 110^o or low inclinations).
2. Pointing accuracy 0.5 deg with knowledge of pointing to 0.2 deg.
3. 36 hr repeat coverage globally in consonance with 1000 km cross-track surface coverage.
4. 10 n.mi coverage required at equator in 3 to 6 months.

Figure 3.4-3 shows the longitudinal advance of the ground track of an orbit approximately every 36 hours. The broken line is the advance per revolution. An orbit design objective is to select an orbit altitude so that the advance is nearly zero in a 36 hour period.

Consecutive orbit passes advance about 3000 km. Since the desired satellite coverage is 1000 km cross range, it is also important that the selected orbit

altitude provide at least two other passes during the 36 hours which cover the area between two initial consecutive passes. Two orbit altitudes which satisfy this and the requirement of nearly zero advance in 36 hours are shown in Figure 3.4-3 as 420 N.Mi (780 km) and 545 N.Mi (1009 km). The basic data presented in this figure is for a spherical earth. Effects of earth's oblateness will increase these altitudes by about 9-11 N.Mi. for the inclinations of 108 to 110 deg.

Figure 3.4-4 shows an example of the surface coverage for an orbit with 120^{o} inclination. Rev 0 starts with an equatorial descending crossing at 180^{o} degrees longitude. One revolution later the orbit will have advanced 26^{o} in longitude (2900 km.). Halfway around the seventh orbit on the ascending equatorial passage, data can be obtained 1000 km. away from the rev. 1 crossing. The fourteenth crossing will provide the data in between rev 0 and rev 7½ to complete the 1000 km swath width coverage. Mapping will be complete in 20½ revolutions (36 hours) and repeat coverage will start. Similar coverage will occur between rev 1 & 2, 2 & 3, etc. Proper adjustment of the orbit period will allow the ascending passage of rev 20 to pass within 10 N.Mi. of the descending passage of rev 0. In approximately 80 days the satellite will achieve the required 10 N.Mi. coverage at the equator.

3.5 ORBIT ACCURACY

Extremely accurate control of the period is required to satisfy the 10 N.Mi. coverage at the equator. Approx ± ½ N.Mi. error in average altitude will result in a deviation of ± 6 N.Mi. in the desired 10 N.Mi. coverage. This will be reflected in either inadequate coverage or repeat coverage that takes longer than the desired 3 to 6 months. Correction of the orbital elements will be required after tracking the satellite for approximately 3 to 5 days and subsequent orbit determination. The orbit trim to correct the errors induced in the launch and injection will require about 30 to 50 FPS. No further orbit trims will be required during a one year mission at these design altitudes.

Both the Atlas F/Burner II and Delta launch vehicles have similar injection errors. The three sigma average altitude error is about 10-15 N.Mi. This will require an orbit trim capability of 30-50 fps. The final stages of the Delta launch system cannot provide the trim impulses of this magnitude with sufficient accuracy so a separate orbit trim system is required. It is described in Section 4.2.2.2. The Burner II control system could be modified to provide the required impulse or the satellite could have its own control system. For an assumed ± 3 N.Mi. tolerance, the requirement of 10 N.Mi. repeat coverage means that the orbit trim system must provide velocity to an accuracy of 0.75 FPS.

3.6 SUN ANGLE HISTORY

For candidate orbit inclinations, Figure 3.6-1 gives histories of the angle between the sun and the orbit plane. The variation for 100 deg inclination is small because it is nearly a sun synchronous orbit. For SEASATS, this is undesirable because on objective is to view the ocean under a variety of seasonal conditions. An inclination of 110^{o} has been chosen for the preliminary baseline design.

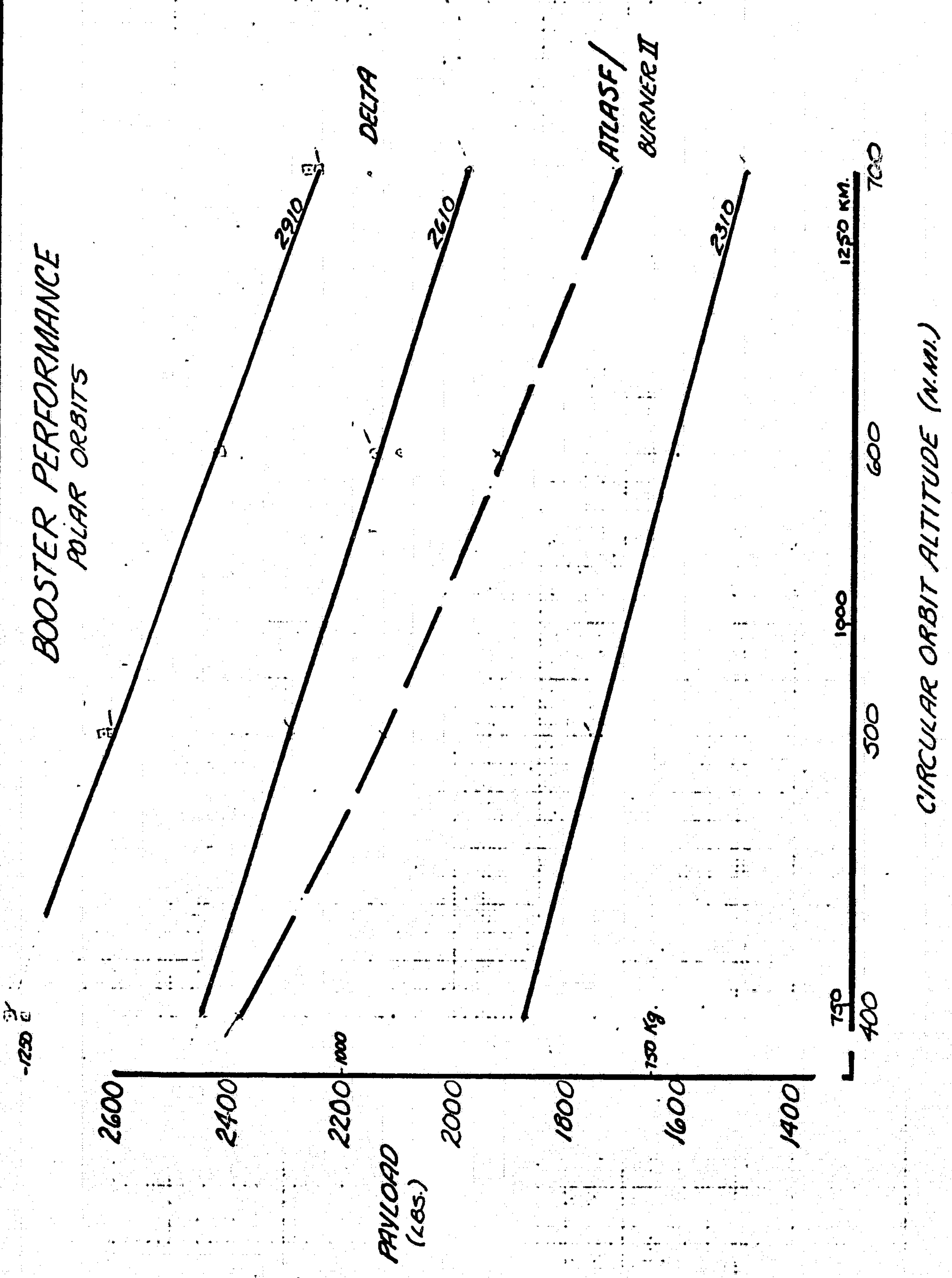
BOOSTER PERFORMANCE
POLAR ORBITS
DELTA
2910
2610
ATLAS F/
BURNER II
2310
PAYLOAD
(LBS.)
2600
2400
2200
2000
1800
1600
1400
-1250
1000
750 Kg.
750
1000
1250 KM.
400
500
600
700
CIRCULAR ORBIT ALTITUDE (N.MI.)

FIGURE 3.4-1

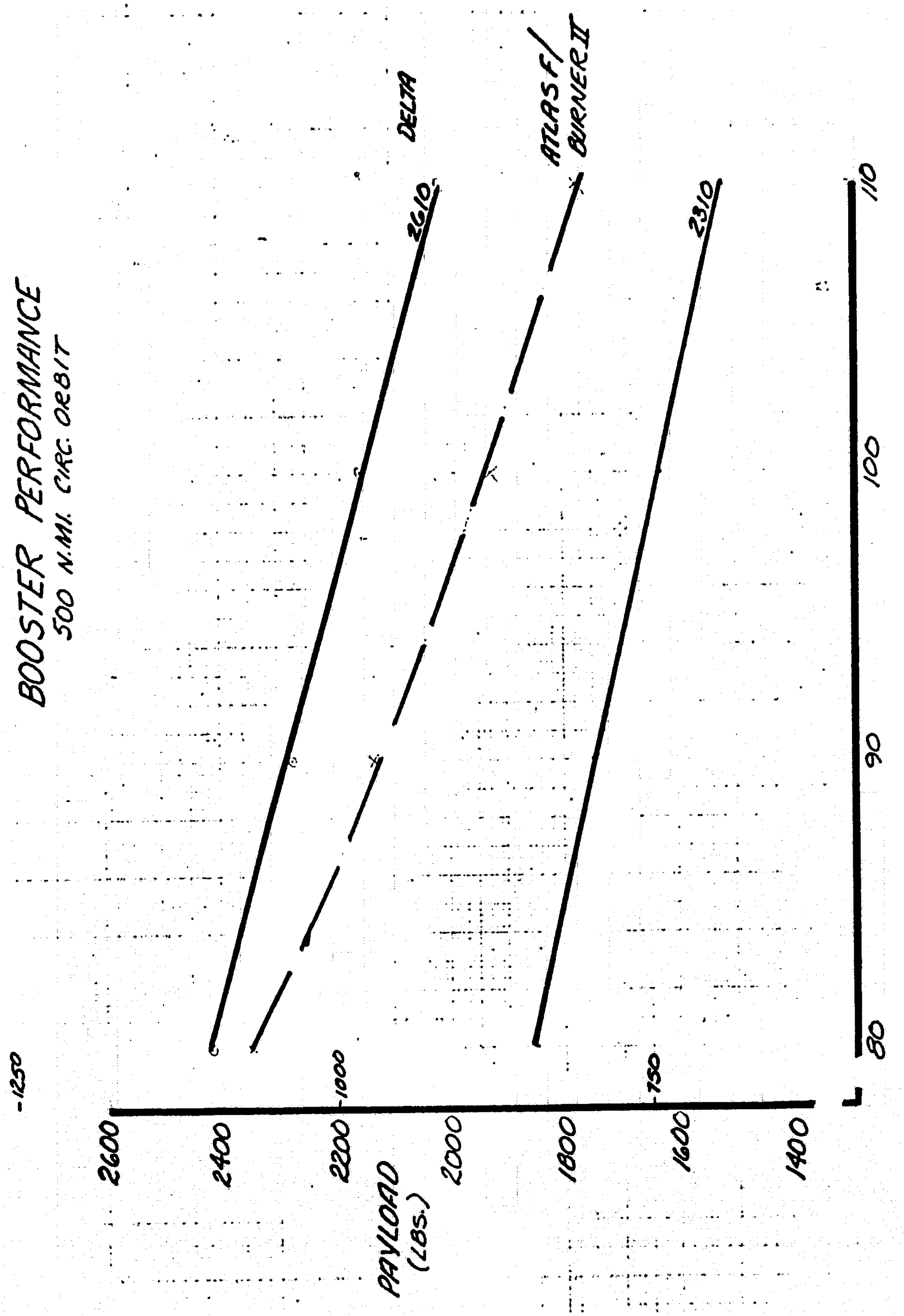
BOOSTER PERFORMANCE
500 N.MI. CIRC. ORBIT
DELTA
ATLAS F/
BURNER II
2610
2310
PAYLOAD
(LBS.)
2600
2400
2200
2000
1800
1600
1400
-1250
-1000
-750
INCLINATION (DEG.)
80
90
100
110

FIGURE 3.4-2

EXAMPLE GR[illegible]ND TRACK

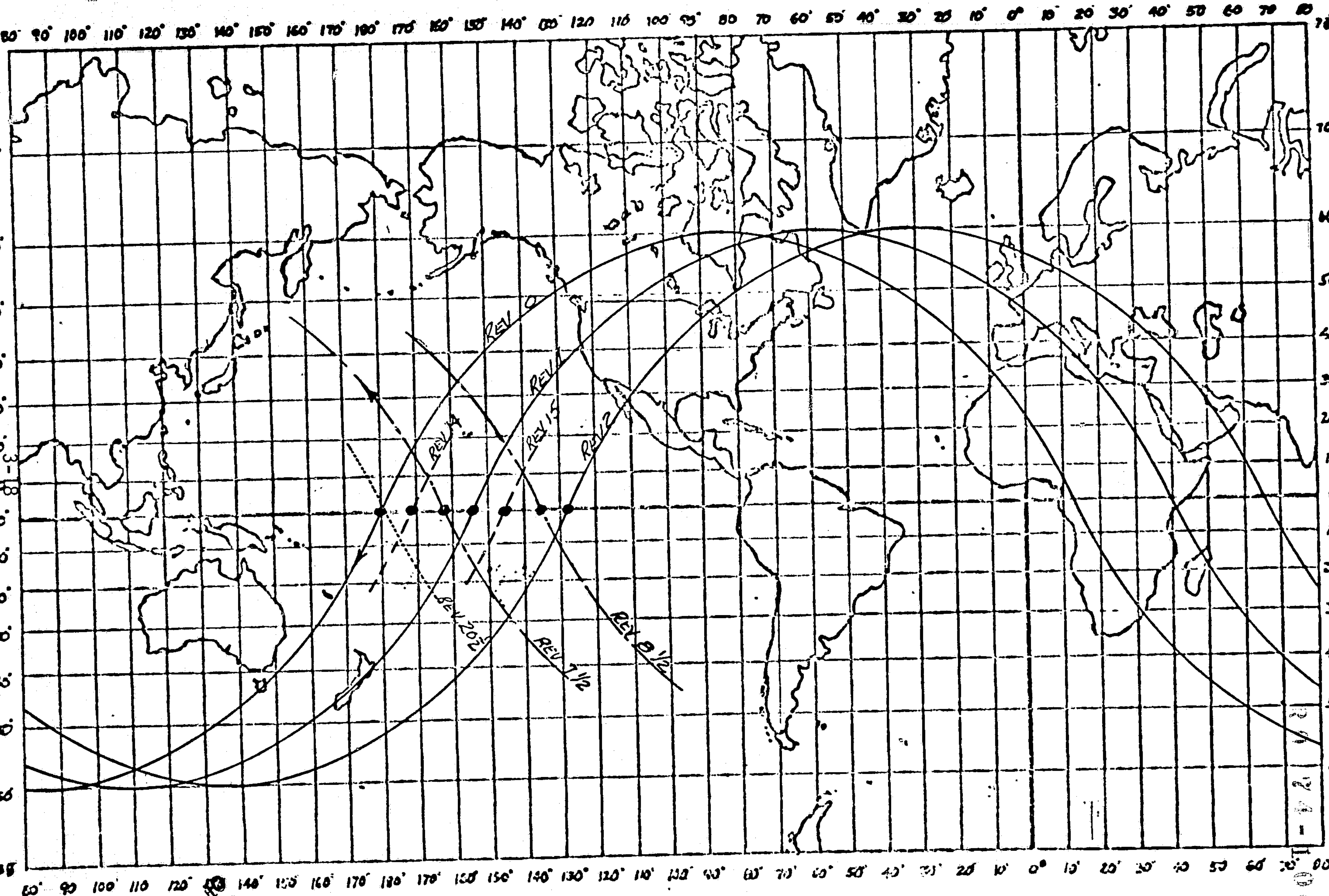

FIGURE 3.4-3

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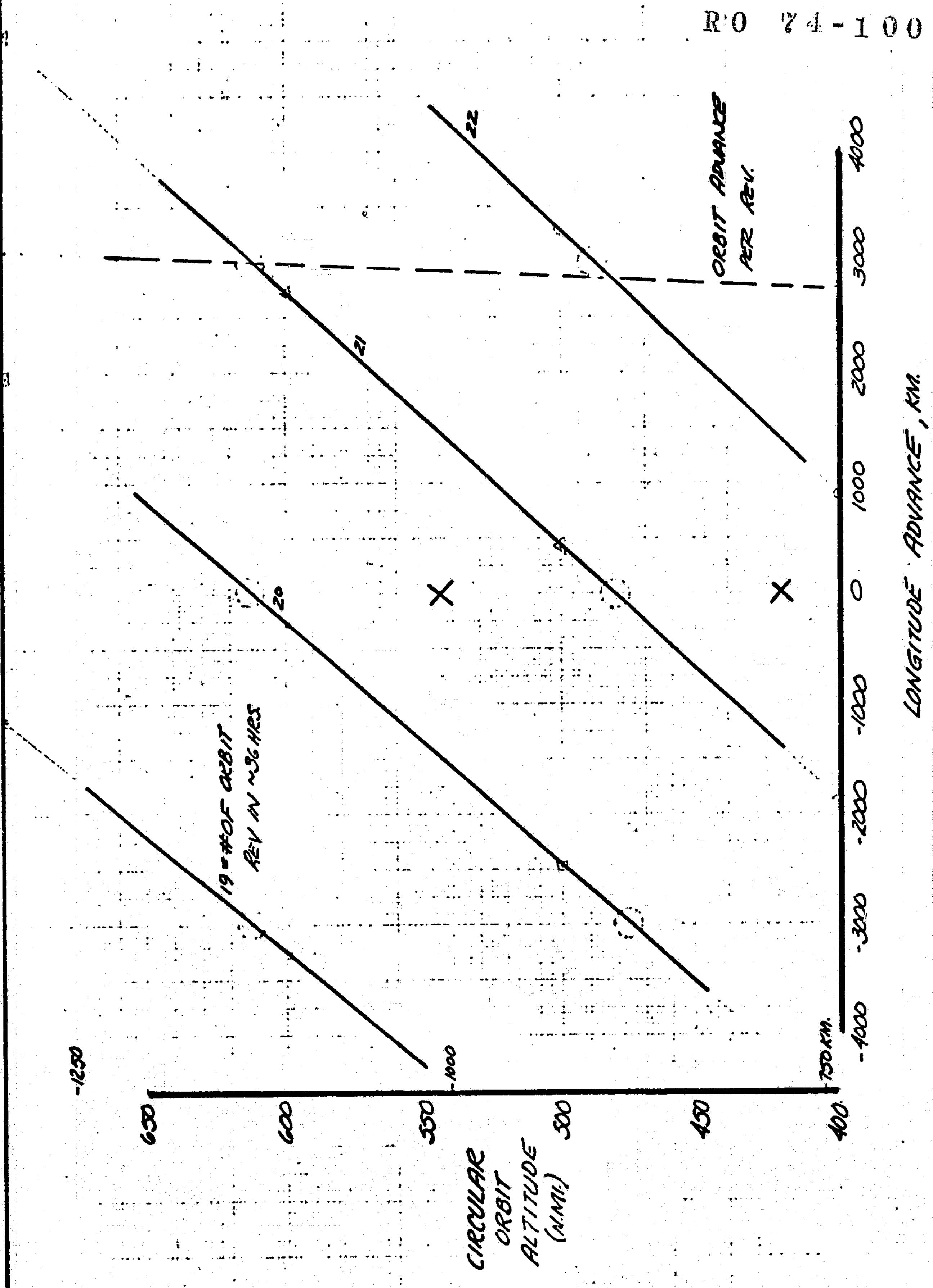

FIGURE 3.4-4

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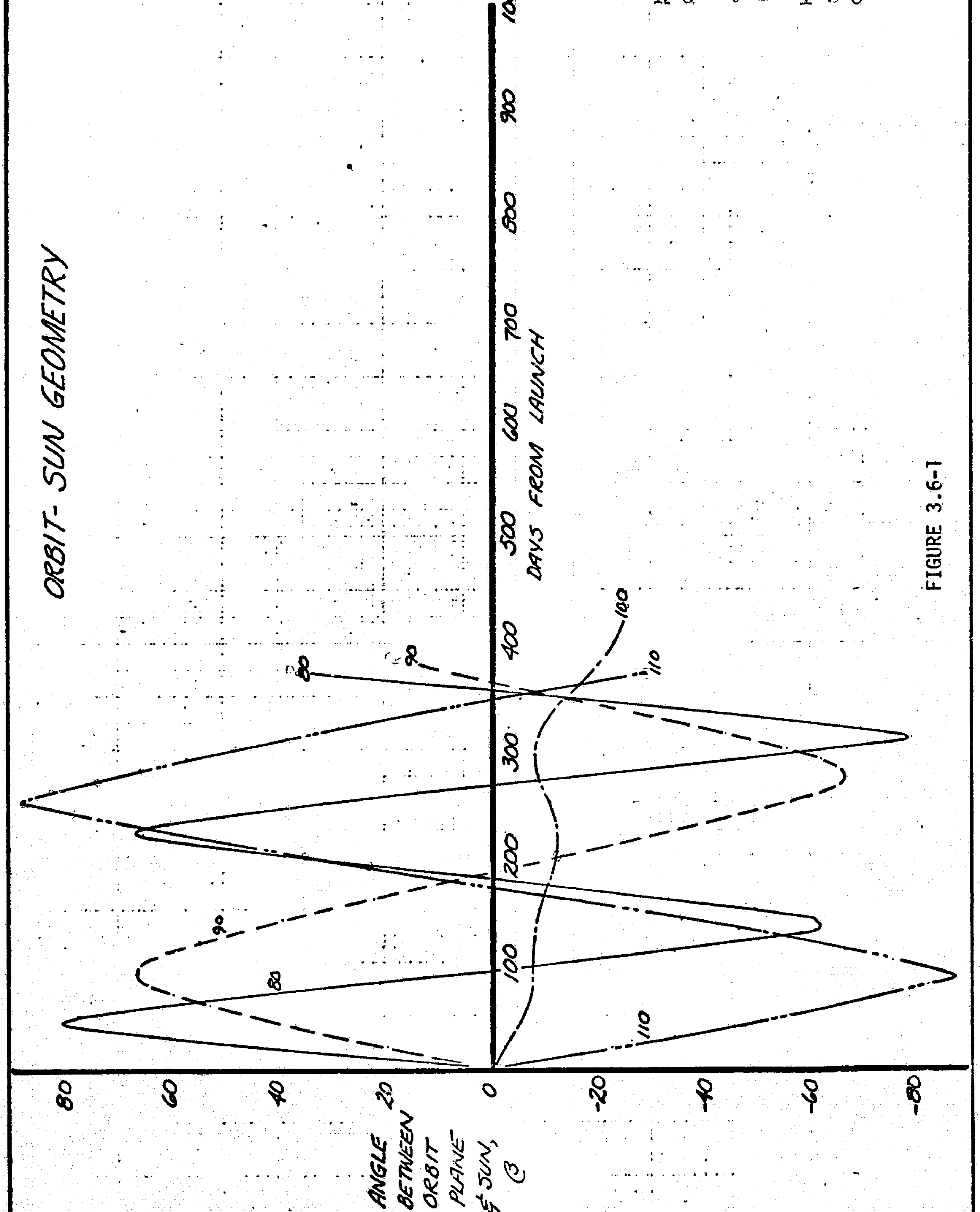


FIGURE 3.6-1

3.7 SPACECRAFT RELIABILITY

A qualitative reliability evaluation of the proposed design indicates a high probability of mission success and subsequent low program risk, based on the design configuration, selected equipment, parts program, technical requirements and program policies.

The design features contributing to high reliability are discussed in detail under the satellite subsystems; in summary the primary features are:

A. Use of flight proven equipment and design concepts.

B. Selection of equipment with demonstrated life.

C. Minimum number of single failure points.

D. Simplicity of the design concept, consistent with the provided redundancy for achievement of Item C above.

E. Ample design margins.

The basic design concept was selected to meet the SEASAT program requirements for a "one shot, one year" mission; these requirements are similar to those successfully met by Boeing for the Air Force STP P72-1 satellite program. Two general concepts were considered: "single string" (with parts screening and traceability beyond the high-rel level) versus extensive use of redundancy. Both concepts have been successfully followed in space programs.

For SEASAT, after a comparative evaluation of the two considered concepts, the selected design with extensive use of redundancy shows definite advantages from technical, economic and program schedule points of view. Following is a summary of items considered during the evaluation.

1. Weight Limitations

   One of the most significant concerns in space programs is satellite weight. This concern, which often overrides other design considerations, making "single string" systems mandatory, is not a governing design factor for SEASAT.

2. Single Point Failures

   The program requirement for the elimination of single point failures can best be met by extensive redundancy. For a long duration mission, where failures are primarily going to be time-dependent (after adequate screening, burn-in and testing), the elimination of single point failures is considered to be a major design objective, which cannot be successfully met by a single thread system. It is recognized that the switching and cross-strapping of redundant systems do introduce some unique failure modes into the satellite, which do not exist in single thread designs. These failure modes will be subject to special considerations during the Failure Mode and Effects Analysis.

3.7 SPACECRAFT RELIABILITY (CONTINUED)

3. Costs and Schedules

The high level screening associated with the single string system, in excess of that normal for high rel parts, can add up to 5% of the total cost of a 100M program*. For a much smaller program, like the SEASAT satellite, the costs would be considerably higher (possibly more than 10%) due to very small and inefficient screening lot sizes.

The single thread system, due to its greater simplicity and greater parts screening, would be expected to have less failures during the test program. It is noted that lack of failures prior to equipment delivery will not normally be reflected by significantly lower equipment costs, as suppliers seldom have firm data relating screening effectiveness to manufacturing test costs.

Although a considerable percentage of the system test discrepancies will be eliminated by the additional screening, the effects of failures on a redundant system are not as critical on schedules (or costs) as for a single thread system. It has been Boeing's experience with the P72-1 program that system tests were seldom delayed by part failures, as testing could normally continue on redundant lines without schedule slides until the repair was effected.

4. Numerical Reliability

It is considered that part failure rates at the completion of the test program prior to launch are no different for the redundant or the single thread concept. In either case, potential problems have been well screened out. The probability of success is an exponential time dependent function that requires redundant paths to achieve acceptably high values (low risk). For the SEASAT "one shot" long mission program, a high probability of success is necessary, as success for a one and only flight is truly an essential mission "requirement".

* Based on JPL data.

3.8 THERMAL CONTROL

The features of the baseline thermal control design include:

- o Independent thermal control for each (payload and bus) module.
- o Electronic equipment mounting platforms nominally held to $\pm 20^{o}$F.
- o A passive louver/radiator system on the payload module for 80-100% duty cycle operation.
- o A heater system which diverts excess power during long periods of equipment shut down, failure, or dormant modes to pertinent local areas in the payload module. No added burden to the power system.
- o A passive louver/radiator system for the bus module holding electronics and control devices to $\pm 20^{o}$F and the battery to 40-70^{o}F during all phases of operation.
- o Orbital temperature stability to $\pm 5^{o}$F.
- o Limiting satellite end-to-end thermal distortions below 0.001 Radians.

In addition, thermal control could be provided for experiments requiring operating temperatures well below the nominal module temperatures with a low power active heater system.

All elements of the baseline thermal control system are flight qualified hardware (Table 3.8.1). They include multilayer insulation blankets, louvers, resistance type heaters, thermostatic switches, black paint, optical solar reflective coatings, solar absorbers, and various thermal control tapes. All items meet, or exceed, a .01% Volatile Condensible Material (VCM) requirement.

Thermal dissipations and weights of major items are shown in Table 3.8.2.

3.8.1 Payload Module

The thermal design for the payload module assumes nominal allowable operating temperatures of 50-90^{o}F, and all equipment on a 80-100% duty cycle.

The sensor electronics and control equipment are mounted internally on the enclosed hexagonal section which makes up the payload module. Passive radiators with controlled solar absorptance-to-emittance ratio (α_S/ϵ_{IR}) coatings will be mounted along the two top sections. In addition, to allow operation at the reduced duty cycle and to make up for small variations in environmental sink temperatures, two 8" X 16" louvers will be mounted on the space facing side of the module. These louvers will modulate approximately 60 watts. Heater assemblies will be placed adjacent to the major power dissipators (altimeter, radiometer and scatterometer electronics), with an additional unit used as a main module heater. These heaters are designated as make-up heaters to be utilized in the event of long term equipment turn-off or major equipment failure. They will be thermostatically controlled with a ground command override capability. The heaters will be resistive type and can operate with raw unconditioned d-c power. These heaters present no extra drain on the power system since they are used only when normal power usage is reduced.

TABLE 3.8.1 EXPERIENCE WITH THERMAL CONTROL EQUIPMENT

ITEM	MANUFACTURER	PREVIOUS USE
Thermal Control Louvers	Northrup	STP 72-1, S3, Mariner Mars '69, Various Classified Programs
Thermostatically Controlled Heaters	Sunstrand Data Control, Inc.	MVM '73, S3, Various Classified Programs
Thermal Coatings/ Coverings -		
o Optical Solar Reflectors	Schjeldahl	Lunar Rover, STP 72-1, SESP 70-1
o Solar Absorbers	Schjeldahl	S3
o Aluminized Kapton/ Mylar	Schjeldahl	S3, STP 72-1, SESP 70-1
o Tapes, Paints	(Various Manuf.)	Lunar Orbiter, S3, SESP 70-1 STP 72-1, MVM '73, etc.

TABLE 3.8.2 ON-ORBIT SPACECRAFT MAJOR EQUIPMENT WEIGHT AND THERMAL DISSIPATION

BUS MODULE EQUIPMENT		WEIGHT/UNIT (POUNDS)	THERMAL POWER/UNIT (WATTS)
POWER			
SHUNT REGULATORS[a]	(4)	2.8	b
BOOST REGULATORS	(4)	10.5	b
BATTERY CHARGERS	(4)	10.5	b
SOLAR ARRAY[a]		108.0	N/A
SOLAR ARRAY DRIVE		11.4	5.5
BATTERY	(12)	10.2	b
RELAY BOX		5.0	0.5
ATTITUDE CONTROL			
SCAN WHEEL		73.0	10.0
ELECTROMAGNETS	(2)	2.0	1.0
CONTROL ELECTRONICS		15.0	8.0
TIMER		3.0	1.0
SUN SENSOR		2.5	0.5
MAGNETOMETER MAGNETOMETER ELECT.		3.8	0.5
TT&C			
RFS SYSTEM		16.0	2.5/13.0[f]
TELEMETRY SUBSYSTEM	(2)	7.0	13.0
COMMAND UNIT		22.0	26.0
TAPE RECORDER	(2)	17.0	20/30[g]
COMMAND RELAY BOX		4.0	0.5
ANTENNA[a]		0.7	N/A
THERMAL CONTROL			
LOUVERS	(4)	0.65	N/A
THERMAL CONTROL ELECT		1.0	0.5

NOTES: a) Located outside bus module

b) Power varies with battery loads

PAYLOAD MODULE EQUIPMENT		WEIGHT/UNIT (POUNDS)	THERMAL POWER/UNIT (WATTS)
SENSORS			
ALTIMETER ANTENNA[c]		10.0	N/A
ALTIMETER ELECT		99.0	90.[d]
ALTIMETER PREPROCESSOR		3.0	10.0
RADIOMETER ANTENNA[c]		15.0	N/A
6.6 GHz CHANNEL	(2)	11.91	8.0
18.0 GHz CHANNEL	(2)	9.48	5.0
22.0 GHz CHANNEL	(2)	9.48	5.0
37.0 GHz CHANNEL	(2)	14.33	8.0
DOPPLER BEACON ANTENNA[c]		1.0	N/A
DOPPLER BEACON ELECT		6.05	5.5
SCATTEROMETER STICKS[c]	(5)	35.0	N/A
SCATTEROMETER ELECT.		85.0	135.[d]
V&IR		20.0	8.0
TLM COMM & TIMING		8.8	5.0
POWER SUPPLY		12.8	14.0
RF SWITCH	(2)	10.0	3.0
THERMAL CONTROL			
LOUVERS	(2)	0.65	N/A
HEATERS	(4)	1.0	e

NOTES (Cont'd): c) Located outside Payload Module
d) Assumed Dissipation
e) As Required - See Text
f) Receive/Transmit
g) Record/Playback

3.8.1.1 Antennas

It is not anticipated that thermal control will be required for the parabolic dishes. Currently, on MVM, a similar antenna, although S-Band, is operating satisfactorily with temperatures ranging from -200 to 250°F across the dish. However, Boeing does have experience in providing temperature control if it becomes necessary. In 1970, an 80-inch dish was developed and tested at Boeing under a full range of sun conditions. A treated kapton tent on the front and a single radiation shield on the back resulted in worst case temperature variations of 140°F. A similar control system for the SEASAT antennas could be utilized if required.

No problems are anticipated with the Doppler beacon antenna which is similar to the flight qualified TT&C antenna.

3.8.2 Bus Module

The bus module thermal control system is completely passive and is similar to the approach used on the highly successful STP 72-1 and the S3 satellite program.

The bus module contains orbit keeping, telemetry, power and housekeeping hardware. The major variations in thermal loads are caused by the power subsystem. The battery packs, boost regulators, and battery chargers are mounted inside the bus module. The shunt regulators are mounted on the first section of the solar panel (which contains no solar cells). In orbits with the largest occultation times (35 minutes) the battery will dissipate approximately 74 watts and the boost regulators approximately 79 watts, during discharge, for a combined orbit average of 54 watts. Four 8" X 16" louvers are used to modulate this excess power. Four are used because 1.) the power system is divided into four modules, and 2.) a narrow temperature control range is desirable for the batteries. Sun shades will be used to protect the bottom louvers from short periods of sunshine, a condition which exists between terminator crossings and sunrise or sunset. The louvers are not qualified for sun facing operation. The side facing louvers will look through the open structure of the rotating solar array storage box. It is estimated that the sun shade and storage box view blockage will reduce the louver performance by approximately 50% to a modulation range of about 5 to 54 watts when operated between 40 and 70°F for the four units.

The thermal control system is augmented with the use of a small (less than one square foot) area covered with solar absorbing material ($\alpha_s/\epsilon_t \gtrsim 20$) on the side of the bus module. This trim area will receive increasing amounts of solar illumination as the orbit angle moves towards a terminator condition. This coincides with a reduction in power system thermal loads due to decreasing battery usage. The net effect is to aid in balancing the bus system thermal loads over the full rnage of orbit angles.

3.8.2 Bus Module (Continued)

The shunt regulators will be mounted on the first section of the solar array (which has no solar cells). Thermal coatings and a variable solar incidence angle will maintain the regulators within operational limits.

The remainder of the bus module equipment will be placed around the interior of the structure. The relatively constant thermal load of these equipments will be dissipated to space through a flat plate radiator along the top two sections. These radiators will be covered with a combination of optical solar reflecting material and black paint to balance the varying external loads over all orbit angles. Internal radiation and conduction will keep the bus module thermal gradients at satisfactorily low levels. The section of the scan wheel that protrudes through the bus (except the mirror assembly) and the remainder of the outside structure will be covered with multilayer blanket insulation. A barrier of blanket insulation may also be placed between the bus and payload module to isolate them from each other if the requirement arises.

The N_2H_4 tanks will be located inside the bus module and held at approximately the nominal bus temperature of 70^oF. The N_2H_4 nozzles (actually one resides in the payload module) will be insulated with a thermal blanket and controlled above 70^oF with heaters operated by thermostatic switches. Each heater will require 0-2 watts of power at a low duty cycle.

3.8.3 Thermal Design Approach

The detailed thermal design will follow the proven approach used in past spacecraft programs such as SESP 70-1, STP 72-1, and S3. This procedure can be summarized as follows:

1. Determine and define all thermal requirements.
2. Develop basic design to meet requirements.
3. Develop a computer math model of the spacecraft and flight environment.
4. Alter design as required by preliminary model results.
5. Perform thermal-vacuum test of spacecraft system using orbital timelines and compare results with model predictions.
6. Update and refine thermal model.
7. Trim thermal control system to final configuration.
8. Prepare flight temperature predictions.

3.8.3.1 Design Tools

The basic design tool will be the thermal math model of the spacecraft. The math model is constructed by dividing the spacecraft into segments called nodes. Each node is assumed to have all mass concentrated at the center of the node and no temperature gradients in the node. A heat balance is then written for each node accounting for all nodal interfaces. The math model is then solved using an IBM 360 digital computer and the Boeing Engineering Thermal Analyzer Program (BETA). The BETA program is a general solution to

3.8.3.1 Design Tools (Continued)

heat transfer solving the matrix of heat balance equations by finite difference for transient state solutions and relaxation for steady state solutions.

The shape factors to be used in the thermal model will be calculated by a computer program using a Monto-Carlo ray tracing technique that accounts for both blockage and surface specularity.

A subroutine used with BETA calculates the energy balance on each external node as a function of spacecraft orbital position; accounting for solar energy, albedo energy, planet radiated energy and deep space. The subroutine provides spacecraft external thermal environment.

3.8.4 Thermal Control Options

In the absence of detailed temperature requirements, the payload baseline design was predicated on experience with typical earth orbiting satellites. The baseline is felt to be the least cost and least risk design. This section is intended to briefly cover and assess the impact of more sophisticated design approaches should more demanding requirements be imposed.

Two systems which would enable closer temperature control ranges, greater orbital stability, a wider choice of individual control ranges and a complete range of duty cycles are 1.) a fully operational heater system, and 2.) a variable control heat pipe system. The relative merits of these systems and the baseline are compared in Table 3.8.3.

3.8.4.1 Fully Operational Heaters

A fully operational heater system normally causes the initial design to be biased at a slightly lower than desired temperature with the heaters being automatically cycled to maintain a set point. This requires solar array power above the nominal S/C bus loads. This typically would increase the power demands to affected units by 10-25%. Heater power required to make up for low duty cycles would not cause extra drain on the power system since it is presently designed for 100% operation. The heaters and heater control electronics would be fully space qualified.

3.8.4.2 Variable Conductance Heat Pipes (VCHP)

The VCHP would be used to regulate the heat flow between individual experiment packages and a space radiator. The range and level of control for each VCHP would be designed according to individual experiment requirements. In this concept, the payload module would be compartmentalized to a degree necessary to isolate packages with different temperature requirements.

Space experience is limited. AMES flew a VCHP experiment on the OAO-C satellite which has worked successfully for over a year. It has demonstrated a capability to stabilize an equipment interface between 19 and 21°C over a wide range of external heat fluxes. A second more advanced system utilizing feed back control is scheduled for flight on the ATS-F satellite. The ground

TABLE 3.8.3 PAYLOAD MODULE THERMAL DESIGN CONCEPT COMPARISONS

THERMAL DESIGN	MAIN FEATURES	SYSTEM COMPLEXITY
(1) Baseline Design - Equipment thermal loads dumped overboard with flat plate radiators with minor variations in internal & external loads modulated by two 8" X 16" louvers; large load fluctuations and/or equipment failures balanced with heater system.	o Passive control during normal operation o Make-up heaters - for abnormal reduced operating levels - automatic with ground command enable capability o No added power requirements o Temperature levels uniform and maintained at better than $\pm 20^{\circ}F$ o Orbital stabiltiy $\pm 5^{\circ}F$	Least
(2) Fully Active Heater System - Same as above except thermally balanced at low end of allowable range. Heater automatically controlled to hold minimum set point.	o Active System o Approximately 10 to 25% added power requirements for all units controlled o Nominal operating temperatures may be different for each experiment. Individual temperatures maintained at $\pm 5^{\circ}F$ o Orbital stability $\pm 5^{\circ}F$	Moderate
(3) Variable Conductance Heat Pipe - Active feed back VCHP used to couple or decouple internal loads from space environment.	o Low power heaters for active feed back control o Wide range on individual equipment operating temperatures o Individual components maintained $\pm 5^{\circ}F$ o Orbital stability $\pm 2^{\circ}F$	Most

3.8.4.2 Variable Conductance Heat Pipes (VCHP)- Continued

test data indicates a very successful design over a wide range of external loads, but still rather small equipment loads (26 watts). This second system with active feed back control would be desirable for the SEASAT because of its higher reliability (less influenced by non-condensible gas generation and a positive control mechanism). Larger capacity or more pipes would be required for the SEASAT altimeter and scatterometer loads.

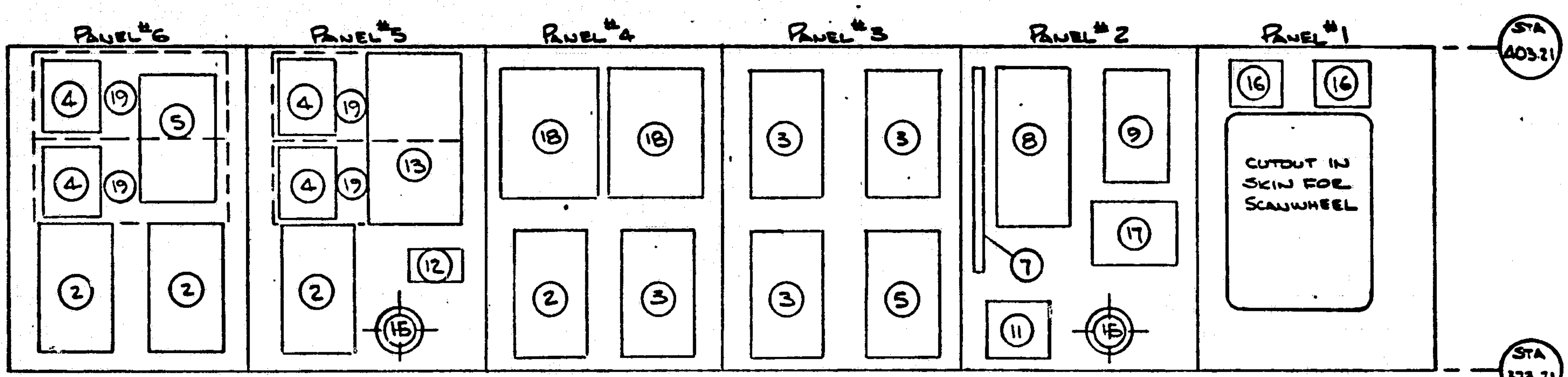
An adequate VCHP design would result in added cost to the program since they are still in the early stages of development. Although the system is complex, it is feasible. Boeing has experinece in analyzing, designing, building, and testing heat pipes and currently has an operating computer program for aiding in the analysis and design of VCHP. Boeing presently has two recognized heat pipe experts on an R and D effort dedicated to heat pipe studies.

4.0 BUS MODULE

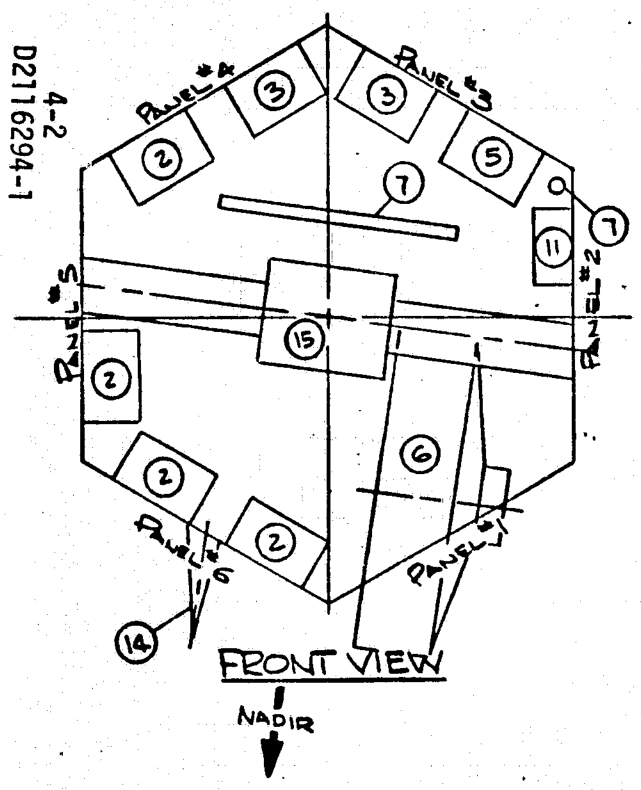
4.1 CONFIGURATION

The proposed "Bus Module" structure is approximately thirty (30) inches long and hexagonal in shape to interface with the existing Burner II stage (STP P72-1). Conventional aluminum structure is used to efficiently meet all requirements for stiffness, strength and equipment support, etc. The module is completely "skinned" and utilizes extruded longerons for the primary load paths. Secondary structural supports are provided for the equipment located within the module. Equipment for the power, attitude control and TT&C subsystems is installed within the module and on the vehicle skin as shown in Figure 4.1.1. Design of the equipment installation considered such items as accessibility for test and replacement, spacecraft center of gravity location and thermal control. Thermal control of the "Bus Module" is accomplished by the use of louvers, paint and blankets. Electrical power is obtained with a 110 sq. ft. single axis articulated solar array. The array consists of two paddles, one on each side of the module inclined at 60° to the Nadir. Each paddle contains six (6) solar panels which are stowed in a box located on the end of the drive shaft. Deployment of the solar array is accomplished by a spring loaded "scissors" mechanism similar to the Lockheed P95 design. Stops and dampers are included in the system to provide positional accuracy and smooth deployment.

The "Bus Module" reflects the STP P72-1 satellite design adapted to accommodate SEASAT equipment installations. All structural concepts and subsystem features are flight proven. The passive thermal control approach has been successfully flown on various space programs and should be just as successful here.



WRAP OUT - INSIDE SKIN



FRONT VIEW

	REF NO	NOMENCLATURE	QTY	SYS
4	1	SHUNT REGULATOR	4	1
	2	BOOST REGULATOR	4	1
	3	BATTERY CHARGER	4	1
	4	BATTERY	4	1
	5	RELAY BOX	1	1
	6	SCANWHEEL	1	2
	7	ELECTROMAGNETS	2	2
	8	CONTROL ELECTRONICS	1	2
	9	TIMER	1	2
4	10	SUN SENSOR	1	2
	11	MAGNETOMETER ELECTRONICS	1	2
	12	MAGNETOMETER SENSOR	1	2
	13	RADIO FREQUENCY SUBSYSTEM	1	3
	14	ANTENNA	1	3
	15	SOLAR ARRAY DRIVE MOTOR & GEAR BOX ASSY	1	1
	16	TELEMETRY SUBSYSTEM	2	3
	17	COMMAND UNIT	1	3
	18	TAPE RECORDER	2	3
	19	LOUVER	4	—

REF NO	NOMENCLATURE	QTY	SYS
20			
21			
22			

4 INSTALLED ON SOLAR ARRAY
3 TT&C SUBSYSTEM
2 ATTITUDE & CONTROL DETERMINATION SUBSYS.
1 POWER SUBSYSTEM

- BUS MODULE, SEASAT BASELINE CONFIGURATION -

SCALE 1:10

FIGURE 4.1-1

AWatry 3/14

74-100

4.2 ATTITUDE CONTROL AND DETERMINATION

The proposed attitude control and determination (AC&D) system has been chosen to make maximum use of flight proven components and to meet the objective of low cost with high reliability. The AC&D system is completely redundant and/or has backup modes of operation except for mechanical elements.

A key element of the system is a large momentum bias wheel that provides high stiffness in roll and yaw with highly predictable dynamics. High stiffness in pitch is obtained autonomously by controlling the torque to the momentum wheel drive motor. Roll and yaw are maintained by supervising the spin axis orientation and applying magnetic correction torques when necessary as determined by ground software.

The system is dynamically similar to the P72-1 and STP S3 satellites built by Boeing. These vehicles were spin stabilized. In SEASAT the spin angular momentum is replaced by the bias wheel momentum but the magnetic torquing is basically the same and some of the sensors are identical. This means that previous experience can be drawn upon, both in design and spacecraft operation, and much of the software already developed can be modified for SEASAT use.

4.2.1 SYSTEM DESCRIPTION

The system is designed to operate in four modes:

- o Acquisition
- o Orbit Trim
- o Orbit Normal Hold
- o Large Angle Maneuver

In orbit normal hold, which is the basic operational mode, the system will maintain attitude within $\pm$ 0.5 deg. 3σ in all axes and permit attitude determination on the ground to within $\pm$ 0.2 deg. 3σ about all axes.

A functional block diagram of the system is shown in Figure 4.2-1. A flight proven momentum bias wheel with a nominal momentum of 300 ft.lb.sec. (392 newton meter sec) provides the high stiffness required to absorb antenna scanning torques and meet control accuracy requirements without active compensation. A redundant horizon crossing indicator mounted on the wheel generates signals that are processed by redundant flight control electronics into roll and pitch errors. Short period pitch control is performed closed-loop on-board by torquing the momentum wheel. Separate windings on the torque motor provide a redundant mode of operation. Both roll and yaw are maintained within limits by supervising the orientation of the wheel spin axis using a dual winding magnetic coil. Roll errors are sent to ground and used by a software program to reconstruct roll and yaw error time histories. Future errors are predicted and an optimum corrective torque schedule determined by the software program. Current levels and switching times are transmitted to the spacecraft and applied on board. The roll/yaw corrections are generated by a coil with its axis nominally parallel to the orbit normal.

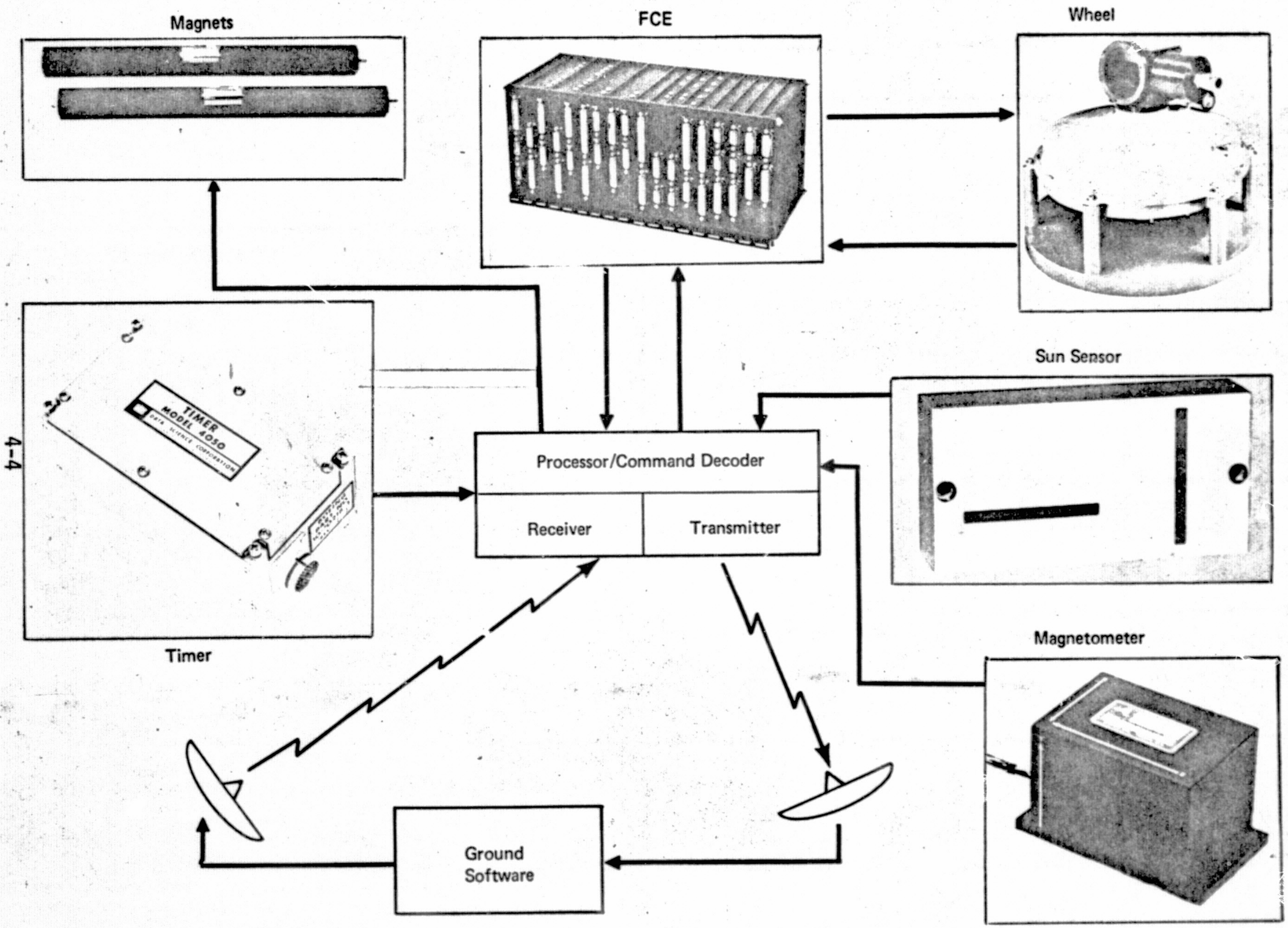


FIGURE 4.2-1: FUNCTIONAL BLOCK DIAGRAM

A second dual winding electromagnet, with its axis along the spacecraft roll axis, is used to desaturate the wheel. Wheel momentum will be maintained within ± 10 ft.lb.sec. (13 newton meter sec) of the nominal bias value. If the wheel momentum approaches a limit the ground station will be informed. The software program will then generate optimum desaturation torque commands.

No appreciable coning motion will be excited during normal operation so the incorporation of a wobble damper is unnecessary. The tight rate coupling which results from the large bias wheel together with flexibility in the structure and solar panel array will combine to provide adequate passive damping.

The attitude control system, in summary, is autonomous in pitch with all sensing and corrective functions performed on board. Spin axis reorientation and wheel desaturation are performed by energizing two electromagnets in accordance with a torquing program generated by ground software. The loop, in effect, is closed on the ground. This approach simplifies the on-board equipment and increases reliability. The response is not as fast as could be provided by an on board autonomous system. The large angular momentum bias, however, makes quick response unnecessary. Experience with the P72-1 and STP S3 programs has proven this approach and demonstrated high accuracy. It is expected that coil command schedules will require updating only infrequently. Figure 4.2-2 shows P72-1 and STP S3 experience and indicates a SEASAT command duty cycle in the neighborhood of 18-20 days.

Accurate knowledge of attitude for all three axes is determined on the ground by a software program that processes pitch and roll data from the scanwheel to compensate for oblate earth and knowledge of altitude from current orbit ephemeris. A sun sensor is included to verify derived yaw data and to enable attitude determination from single station contacts. A three-axis magnetometer provides attitude data during large angle maneuvers when the earth is outside the scanwheel detector field of view. The magnetometer also provides a backup mode in the event of horizon sensor failure. The software program will be similar to and adapted from the programs already developed for the P72-1 and STP S3 satellites.

4.2.2 CONTROL MODES

The Burner II space launch vehicle will place the satellite in orbit. Just prior to separation the three axis control system will orient the payload so that it is within ± 10 degrees of its desired operating attitude. Prior to separation the bias wheel will be partially spun up. This serves two purposes - it provides stability thereby eliminating any possibility of tumbling and enables attitude determination immediately after separation.

4.2.2.1 Acquisition

In the acquisition mode the ground software will generate coil torquing commands to accelerate the wheel to its nominal operating speed and reduce orientation errors to within ± 0.5 deg. Ground station contact with the spacecraft every two to four orbits will be required. The process will take several orbits to complete. During this time an accurate orbit ephemeris is obtained and payload checkout can be accomplished.

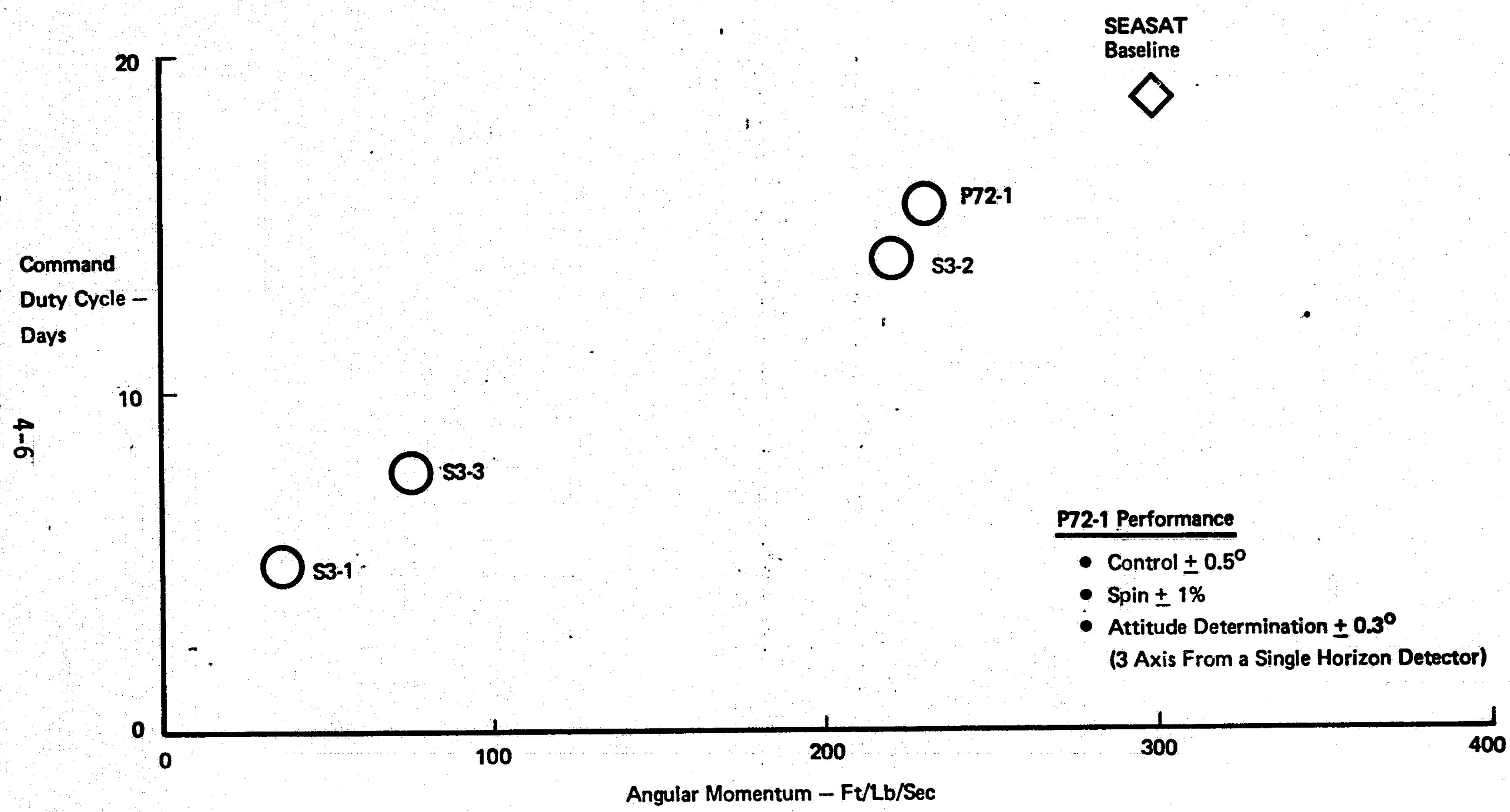


FIGURE 4.2-2: COMMAND DUTY CYCLE

The baseline system takes advantage of the three axis control capability of the Burner II launch vehicle to achieve coarse acquisition leaving fine acquisition to the spacecraft. The AC&D system however has full all-attitude acquisition capability limited only by telemetry, thermal and power constraints. The magnetometer, earth sensor and sun sensor, in combination, provide all necessary attitude data and the control coils need only the appropriate commands to torque the vehicle to any desired attitude. Software programs to perform these maneuvers have been developed for the P72-1 and STP S3 vehicles and flight experience in their use gained with P72-1.

4.2.2.2 Orbit Trim

SEASAT orbit trim is provided by a monopropellant hydrazine propulsion system. The system, shown schematically in Figure 4.2-3, has a single 6 Al-4V propellant tank with EPT-10 positive expulsion bladder, and two low thrust hydrazine engines aligned fore and aft along the roll axis to provide velocity correction as required after separation from the boost system. All components are flight proven designs and are now or have recently been in production.

Initial orbit trim requirements are 30-50 ft/sec (9.15-15.24 m/sec) ± 0.75 ft/sec (0.23 m/sec). A thrust level of 0.5 lb_f (2.22N) permits the wheel torquer to counter any misalignment torques in pitch. Limiting each burn time to not more than 210 sec. will ensure attitude errors of less than 2.0 degrees. An intermittent ΔV application has the advantage of allowing attitude error correction between pulses and also affords an opportunity to update the ephemeris. As in the acquisition mode frequent ground contacts will be necessary. As experience accumulates over several ΔV applications, the thrust offsets will become defined. This knowledge will simplify and may speed up the corrections applied in successive burns.

Table 4.2-1 is a weight statement for the orbit trim propulsion system. Propellant quantity was based upon providing 50 ft/sec (15.24 m/sec) ΔV to a 2000 lb_m (910 Kg) spacecraft with -2σ thruster performance.

The propellant tank is a 16.7 inch (42.4 cm) diameter spherical titanium shell with integral ethylene propylene terpolymer (EPT-10) diaphragm. The weight statement is based upon the ATS F&G/MVM'73 propellant tanks with three lugs located near the tank girth as mounting points. Pressurization is by gaseous nitrogen in a blowdown mode. Initial pressure is approximately 300 lb_f/in^2 (2.07×10^6 N/m^2). Blowdown is only 1.2 to 1 for the tank and propellant requirements given above. An existing smaller tank would be adequate at the lower study ΔV requirement of 30 ft/sec (9.15 m/sec) or at 50 ft/sec (15.24 m/sec) if spacecraft weight were at the lower level of 1500 pounds (681 Kg). The smaller tank is adequate only for initial orbit trim ΔV with the restricted conditions noted while the tank selected has adequate volume to handle maximum orbit trim requirements and added propellant for more than two years station keeping if desired. Propellant tank thermal control is passive.

The orbit trim thruster assemblies are a nominal 0.5 lb_f (2.22 N) thrust design including a nozzle, decomposition chamber containing Shell 405 catalyst and an integrally mounted series-redundant, soft-seat solenoid valve. Leakage past the valve seat is the most probable thruster failure mode. The dual valve seat provides redundancy against that failure,protecting against uncontrolled impulse

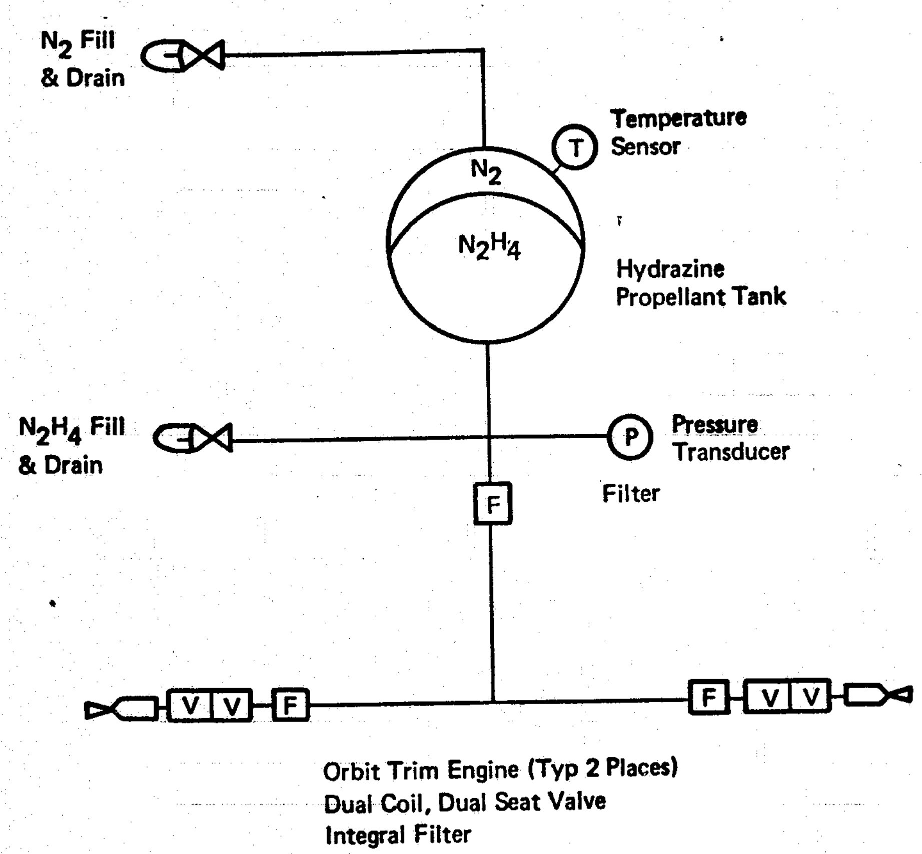


FIGURE 4.2-3: ON-BOARD PROPULSION SYSTEM SCHEMATIC

from the orbit trim system. Total burn time for one engine to provide the maximum ΔV trim is approximately 7000 seconds. Relatively few thruster starts are required. Three manufacturers have qualified and flight proven thruster designs potentially applicable to SEASAT orbit trim requirements. Valve and chamber heaters may be required to prevent propellant freezing and insure smooth thruster starts. It is anticipated heater power requirements will be less than 3 watts per thruster. No heater power would be required after completion of orbit trim unless the system is also to be used for station keeping.

TABLE 4.2-1

ORBIT TRIM SUBSYSTEM WEIGHTS

Item	Weight lb_m	Weight Kg
N_2H_4 Tank	10.2	4.64
N_2H_4 Thrusters (2)	1.8	0.82
Fill & Drain Valves (2)	0.5	0.23
Line Filter	0.5	0.23
Pressure Transducer	0.2	0.09
Temperature Sensor	0.1	0.05
Propellant Lines	1.5	0.68
Tank & Thruster Mounts	1.5	0.68
N_2H_4	16.0	7.27
N_2	1.6	0.73
Total Weight (Loaded)	33.9	15.42

Other subsystem components are:

- External cap seal manually operated fill and vent valves. One each for propellant fueling/defueling and propellant tank pressurization.
- A stacked disc type propellant line filter.
- Pressure and temperature instrumentation for propellant servicing data and to allow estimation of flight performance.
- Propellant lines and components are joined by induction-brazed fittings.

Ground checkout and servicing systems for the orbit trim subsystem are assumed to be available GFE. The pneumatic and electrical checkout equipment, vacuum

pump, propellant transfer equipment and scales from the MVM'73 program are applicable and assumed available to this program.

4.2.2.3 Orbit Normal Hold

Once acquisition and initial orbit trim have been accomplished the system will move into the orbit normal hold mode. It will stay in this mode for the remainder of its useful life except for 180 degree maneuvers and possibly an additional period of station keeping in the orbit adjustment mode if this should become necessary. It is expected that once initial transients have damped down and the system settles into a regular pattern of behavior, updates to the coil torquing schedule will be necessary only once every 18-20 days maximum.

Pitch control is performed on board using a conventional closed loop system in which pitch errors, sensed by the scanwheel, are processed to generate wheel torques. Roll and yaw errors are corrected by a precession coil energized in accordance with a schedule determined by ground software.

A constant precession coil current will lead to a rotation of the vehicle in right ascension while a current that changes polarity at the equator and closest approach to the poles leads to a rotation in declination. Any desired rotation can thus be achieved by commanding a combination of constant and alternating coil currents. Moreover the combination results in only two coil current levels which are applied alternately.

The torquing schedule is implemented as shown in Figure 4.2-4. The two current levels, designated P_1 and P_2 and their polarities, are transmitted to the satellite, processed by the command decoder and stored in two 7-bit registers. The currents are applied to the coil at times determined by the orbit sequencer. These times correspond to passage over the equator and the nearest approach to the poles. Since the orbit is circular these time intervals will be equal. Synchronization is maintained by a synch. drive signal derived from the sequencer clock.

Momentum wheel desaturation is achieved in much the same way. The current is switched only at the equator in this case to produce an accumulating torque. Two levels only are provided - a low level for normal desaturation and a high level for rapid spin/despin of the wheel during acquisition and large maneuver sequences. On/off and spin up/down signals complete the list of commands.

4.2.2.4 Large Angle Maneuver

Twice in the orbital lifetime a 180 degree yaw turn will become necessary. This is required since the solar panel configuration will only operate with the sun on one side of the satellite. When the sun moves from one orbit hemisphere to the other, due to orbit precession, the yaw turn is required.

The maneuver will be achieved with the torquing coils on ground command. Analysis shows that maneuver time can be reduced by partially spinning down the wheel, performing the 180° precession and then accelerating the wheel back up to speed again. The total time for the operation is about 5 days of which

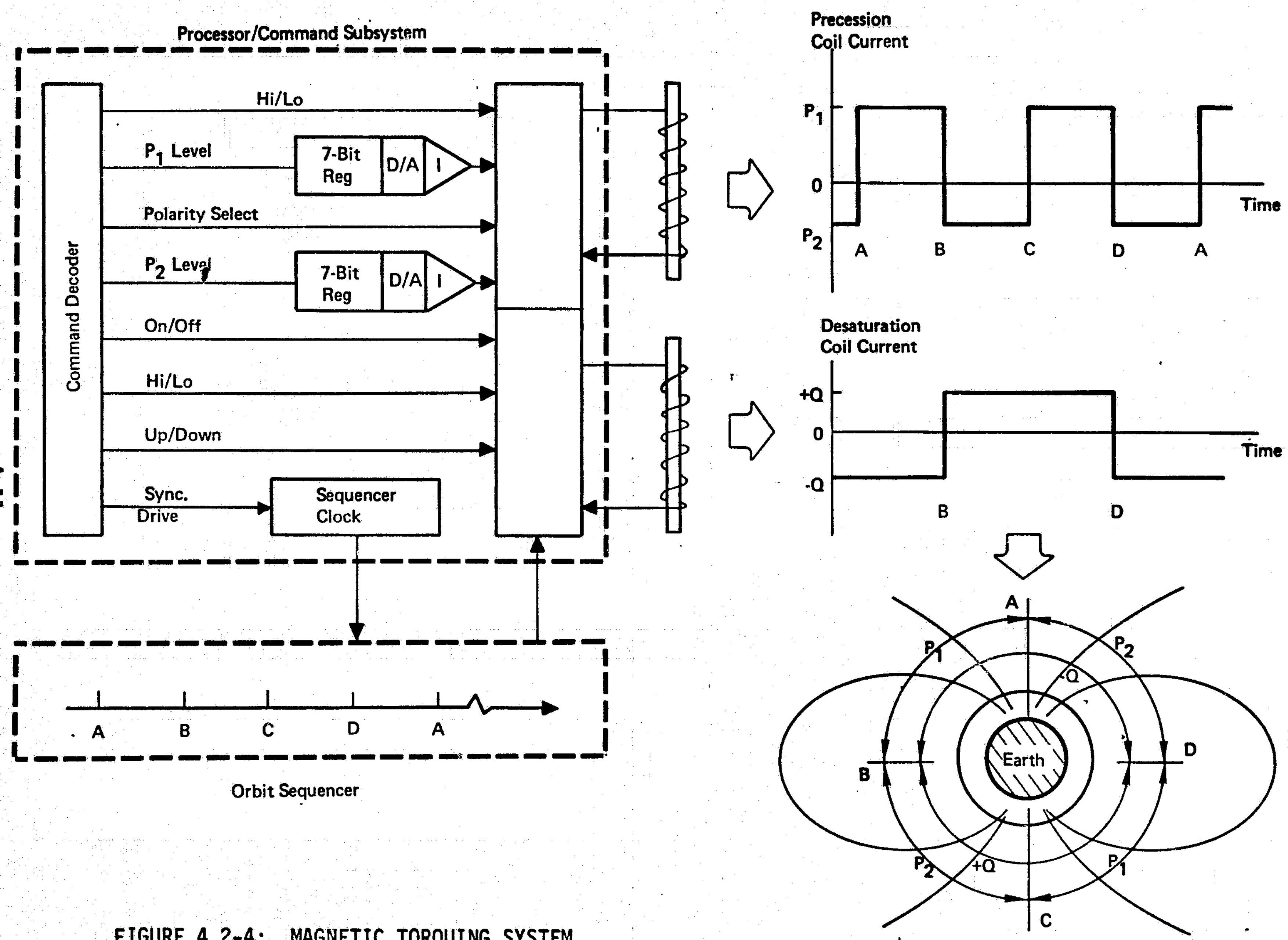


FIGURE 4.2-4: MAGNETIC TORQUING SYSTEM

3 1/3 days are spent precessing the vehicle. Normal operation can be conducted during the spin down and spin up phases with a possible small degradation in accuracy. Similar maneuvers are part of the STP S3 flight plan and the software for control during maneuver has been developed.

Changes in control gains and signs become necessary to operate the spacecraft after 180 degree yaw maneuvers. These are incorporated in the ground software. The autonomous pitch control loop will not require modification.

4.2.3 COMPONENT DESCRIPTION

The AC&D system components and their main physical characteristics are listed in Table 4.2-2.

The momentum bias system with magnetic control has seen a number of applications and thus has a Functional Classification 1. The system has flown on ITOS and SAS. Dynamically equivalent spinning vehicles that have used magnetic control include TIROS and P72-1.

Hardware design heritage is shown in Table 4.2-3. All components have flown on other spacecraft (Classification 1b) except the horizon scanner. Ithaco has adapted several Bendix and Sperry wheels to carry various scanner-detector assemblies. The scanner proposed for SEASAT is under development.

4.2.3.1 Scanwheel

The Model 35 Conical Scanwheel is an attitude control system component that combines the following functions into a single integrated piece of hardware:

Pitch and Roll Attitude Determination ($< 0.2^{o}$)
Momentum Bias
Closed Loop Pitch Control ($< 0.5^{o}$)

This scanwheel is a modification of the Planar Scanwheel, a joint development of Sperry Flight Systems of Phoenix, Arizona, and ITHACO, Inc. of Ithaca, New York. It combines a Sperry Momentum/Reaction Wheel with an ITHACO Infrared Horizon Scanner. A planar scanwheel has been successfully qualified to GPS/VP sine and random vibration levels, which are significantly more severe than those for SEASAT.

The horizon scanner utilizes the momentum wheel shaft to rotate the single moving element, a berylium mirror. This mirror is tilted to deflect the field of view and generate a conical scan pattern. Pitch attitude is computed by comparing the time position of the earth pulse relative to a magnetic pickup reference pulse that indicates the position of the mirror. Roll attitude is computed by comparing the length of the scan paths.

A magnetic pickup is so positioned that a pulse occurs when the detector image is aligned with the nominal local vertical. This pulse will occur in the center of the earth pulse if there is no attitude error around the pitch axis.

Pitch axis errors are then computed by comparing the angular interval between the leading horizon and the reference pulse to the interval between the

TABLE 4.2-2

COMPONENT	MANUFACTURER	SIZE	POWER	WEIGHT
Scanwheel	Sperry/Ithaco	26" D. x 9"	10W	76 lb.
Electromagnets (2)	Boeing	1" D. x 19" ea.	1W ea. av.	2 lb. ea.
Sun Sensor	Adcole	1" x 1.5" x 2.5" 1.5" x 2.25" x 3.25"	0.5W	1 lb.
Sequencer	Data Science	3.5" x 6" x 11"	1W	3 lb.
Magnetometer	Schonstedt	3" x 3" x 5" 3" x 5.5" x 6.5"	0.5W	4 lb.
Flight Control Electronics	Ithaco	6" x 6" x 8"	5W	8 lb.

AC&D COMPONENT CHARACTERISTICS

TABLE 4.2-3

COMPONENT	CLASSIFICATION		
	MFGR	HERITAGE	COMMENTS
Scanwheel			
Momentum Wheel	Sperry	1b., Classified S/C	Without shaft extension to mate scanner. Shaft extension developed and qualified for GPS.
Horizon Scanner	Ithaco	3, Nimbus/ERTS, ELMS, P72-2	Adapt low altitude scanner to Sperry wheel and provide redundant detector.
Electromagnets	Boeing	1b., P72-1, S3	-----
Sun Sensor	Adcole	1b., STP S3, OAO	STP S3 simplification using Gray Code and single slit/axis incorporated in 2 axis (OAO type) sensor.
Sequencer	Data Science	1b., STP S3	-----
Magnetometer	Schonstedt	1b., P72-1	-----
Flight Control Electronics	Adcole	1b., P72-2	Cost effective approach proposed without repacking to save weight. Improvements to increase accuracy by circuit mods proposed to: 1) Add tach feedback for wheel speed Control 2) Revise Horizon Scanner Data Processing

HARDWARE DESIGN HERITAGE

reference pulse and the trailing horizon. Attitude errors around the roll axis cause the earth pulse width to change. For a circular orbit the earth pulse can be compared to a predetermined reference level.

The scanner optical system uses flight proven techniques and employs a thin film filter on a Ge substrate to define the 14-16μm IR passband, and a Ge lens to focus the incident energy on the mylar immersed thermistor bolometer.

The preamplifier assembly is contained within the scanner assembly that is attached to the wheel. This electronics module is packaged around the optical barrel and contains the following circuits:

- o Bolometer bias circuit and power supply filter
- o Preamplifier
- o Peaking amplifier which compensates for the thermal lag of the bolometer
- o Sun limiter which prevents saturation of the amplifier when the sun is within the scanner field of view.

The mechanical design of the mirror provides for a balanced assembly which may be rotationally adjusted for alignment with the tachometer pulses. The housing mounts firmly to the wheel housing and is pinned in place for increased accuracy. The window is a thin plastic film which will withstand a 1.0 atmosphere pressure differential. This window is designed for ground test and storage protection and is removed at launch.

Scanner Characteristics

- o Instantaneous field of view 2 x 2^o
- o Scan limits about pitch axis = $\pm$ 90^o
- o Power consumption = 0.5W max total from $\pm$ 10, and +115V lines (preamp only)
- o Accuracy (exclusive of orbit eccentricity and oblateness) $< \pm .2^o$ in pitch and roll (3σ)
- o Signal output = 1V/degree, saturating at $\pm$ 8.5V.

Attitude Determination Accuracy

There are many factors that affect the accuracy of the pitch and roll information derived from horizon scanners. The following discussion describes the error sources and gives some feel for the magnitude of the attitude determination errors that result.

Horizon Effects - Extensive analyses have shown that the optimum optical passband to minimize the effect of horizon height variations is a narrow interval centered at about 15 microns. With this optical passband, and

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using an optimized horizon locator technique the effective horizon height varies by about 3 Km and is predictable to about 1 Km. The predictable effects include seasonal, latitude, and longitude effects.

Alignment and Calibration Effects - The conical scanner optics are normally aligned and calibrated to the base plate with an absolute null offset of $\pm .1^{\circ}$ maximum with respect to the SCANWHEEL mounting surface.

Oblateness Effects - The earth is not round, and this leads to some confusion as to which direction is really "down." This effect is entirely predictable, being of the order of $\pm .1^{\circ}$ peak in a 1500 Km circular orbit. For lower orbits, the error increases, and for synchronous equatorial orbits is negligible.

Orbit Eccentricity Effects - There will be no eccentricity error in the axis parallel to the SCANWHEEL spin axis. The eccentricity error in the axis perpendicular to the spin axis of a single conical scanner is predictable from the geometry. For example, the eccentricity error for a spacecraft in a nominal 650 Km orbit will be about $.3^{\circ}$ per 10 Km change in altitude.

Sun-Moon Effects - Considerable attention has been given to minimizing the effect of the sun. The effect of presence of the sun anywhere, even directly in the field of view, can be limited to an attitude determination error of no greater than $.1^{\circ}$, and in most circumstances considerably less.

Redundancy

Optical redundancy will be implemented through the use of a dual flake bolometer. Each flake will be biased and processed separately by redundant electronics. This form of redundancy has been used on the ELMS flight program. The optical elements other than the bolometer have such a small failure rate that they need no redundancy.

The use of redundant motors is not suggested to meet the program requirements. Instead, a design approach is proposed that has been successfully employed on P72-2 and ELMS programs to enhance motor reliability and as a by-product ease the redundancy implementation of motor driving. Under this scheme each motor is wound with two non-bifilar windings per phase. This winding technique minimizes mutual coupling between the windings. The motor may be driven with less than a 15% decrease in torque from only one pair of windings (one from each phase). This type of motor construction in no significant way alters the qualification status or major mechanical properties of the motor. It also has the added advantages of cost and weight savings.

Qualification Status

The Type 35c Conical Scanwheel is derived from the Model 45P Planar Scanwheel. The optics assembly and preamp are identical in the two scanwheels. This

assembly is attached to the scanner housing at the end rather than on the side as in the planar scanwheel. Both the housing and mirror mount interfaces are identical in the two scanwheels. The planar scanwheel has been successfully qualified to GPS/VP sine and random vibration levels.

All optical techniques employed in these scanwheels are derived from flight proven designs.

4.2.3.2 Flight Control Electronics (FCE)

The FCE for the scanwheel consists of 10 modular cards with each function duplicated to eliminate single point failures. These functions are redundantly performed by the following cards:

		Quantity
A)	Power Supplies	(2)
B)	Motor Drivers	(2)
C)	Signal Processors	(2)
D)	Reference Processors	(2)
E)	Power Switching	(1)
F)	Pitch/Roll Control	(1)
	TOTAL	10

The cards are stacked together as shown in Figure 4.2-1. This modular construction offers maximum flexibility and commonality with previous flight programs. In particular, similar stacks have been used on P72-2 (18 card stack), ELMS (17 card stack) and SAS-C (3 card stack) flight programs. Engineering hardware using these assemblies has also been built for IUE and SATS (GSFC).

The FCE for SEASAT will use power supply and motor driver modules which are identical to those on the above programs. Similarities in other cards permit accurate estimates of size and weight for the FCE and eliminate technical risk.

The FCE performs the following functions:

a) Processing of horizon scanner earth pulses to produce pitch and roll position error signals.

b) Supply power to the horizon scanner assembly.

c) Process the pitch error signal according to the pitch control law and drive the momentum wheel motor driver.

d) Provide tachometer information, a minimum speed tach loop and a wheel unloading signal to the magnetic assembly.

e) Provide telemetry outputs for the functions provided.

Redundancy Implementation

The redundancy philosophy follows closely to that successfully used on the P72-2 and ELMS Programs. The configuration minimizes the switching of signal leads and performs most redundancy switching by the removal of power from the offending card.

The power supplies, consisting of two separate DC to DC converters, will have their output voltages strapped together to permit operation on either or both supplies. The supplies are isolated from each other by redundant diodes. Both supplies will be capable of sustaining shorted outputs indefinitely and will return to normal regulation when the failed unit is disabled by ground command.

Functional Descriptions

The technical approach utilizes flight qualified proven designs wherever possible. The performance and environmental requirements of the FCE will be met well within the limits imposed by the preliminary specification with the exception of operating power. The text below will discuss briefly each circuit function:

Power Supplies - The DC to DC converters for voltage conversion and regulation are similar to those flown on Nimbus/ERTS and used on P72-2 and ELMS vehicles. Parallel strapping of voltages for redundancy provides for maximum redundant mode operation with minimal switching. The power supply provides ± 10V at up to .75A, + 115V for the HSA, + 5V for TTL logic and a regulated ± 5.0V reference voltage. Output regulation is within ± 5% on the ± 10V and + 5V lines and ± 0.5% on the ± 5V reference supply. The +28V bus is preregulated and filtered on the primary side of the converter.

The power supply produces the sawtooth for the switching regulator in the motor driver, and generates the two phase clocks.

Reference and Signal Processors - The planar scanwheel utilizes a type of synchronous detection to process the earth signals at the high speeds present with the Model 35 momentum wheels.

The circuitry performs 3 functions, which are described below.

a) Tachometer Processor: The tachometer processor decodes the multiplexed tach pulses which provide information for the location of the center of the scan and blanking at the extreme edges of the scan. The tachometer signal is delayed by an amount precisely equal to the total signal channel delay to eliminate errors due to wheel speed fluctuations.

b) Reference Waveform Generator: A "V" shaped reference waveform is produced from the decoded tach signal. This waveform provides a conversion between scan degrees and voltage that is not speed dependent.

c) Signal Processor: The earth horizon crossings are determined by a form of electronic edge tracking. Each horizon crossing is sampled in 2 adjacent intervals during each scan. The sampled signals are combined and fed to a precision integrator. Only when the sampling occurs on the earth pulse edge does this combination of sampled signals equal zero, which allows the integrator to remain at a steady value. The integrator DC output voltage when compared to the reference waveform is used to gate the sampling circuits. The integrator DC voltage levels are proportional to the angular position of the earth horizon crossings. To reduce the complexity some circuits are used to process both horizon crossings by signal multiplexing.

The two integrator output signals and a DC bias are algebraically combined to give pitch and roll error signals.

The output in pitch and roll will be approximately zero (null) with the loss of the earth signal.

Pitch Computer - The pitch loop electronics will consist of a minimum speed tach loop, the control law (lead/lag) circuit, and the wheel unloading command circuit.

Power Switching Card - The power distribution and cross strapping of redundant circuitry will be accomplished by discrete ground commands via flight proven latching relays. The following discrete commands will be used on this card:

All redundancy switching

All power on/off switching

All major mode changes

Two similar cards of this nature exist on P72-2, ELMS and Nimbus/ERTS.

Motor Drivers - The proposed reaction wheel driver features a switching regulator for smooth, linear torque control and a four transistor bridge to drive the motor windings. The shaped sawtooth waveform from the power supply is used to pulse width modulate the switching regulator which, in turn, produces a DC output voltage proportional to the square of the input torque command magnitude. This voltage is switched by two 4 transistor bridges across the motor drive windings. The torque command polarity determines whether Phase A leads or lags Phase B by 90 degrees.

These drivers are essentially identical to the P72-2, ELMS, and SAS-C spacecraft drivers. The bridge portion of these drivers has successfully performed in orbit on numerous Nimbus/ERTS spacecraft. The proposed method yields driver efficiencies of better than 90% under all operating conditions.

A tach circuit similar to that used on Nimbus/ERTS will be used to measure wheel speed. The decoded tach pulses are fed to a one shot whose output is filtered.

The magnitude of the filter output voltage provides speed information.

Qualification Status

This modular format has been successfully subjected to environmental tests on flight programs including vibration, temperature, shock, and EMI on P72-2. It is anticipated that the FCE assembly will offer approximately the same performance with these environments as the P72-2 assembly.

4.2.3.3 Electromagnets

Boeing has built electromagnets for the STP S3 satellites. Similar units are proposed for SEASAT. The two magnets, which are identical, will each have two windings with each winding separately controlled and producing up to 75000 pole cm. There will thus be redundant capability in each axis. Both windings can be energized together to provide increased torque and faster response in the maneuver and station keeping modes. In the acquisition and orbit normal hold modes a linear ± 35000 pole cm. range will be utilized to provide fine resolution.

4.2.3.4 Sun Sensor

The sun sensor supplements the earth sensor data. The earth sensor cannot measure yaw although yaw can be precisely determined in the Kalman filter from roll measurements. The sun sensor essentially verifies the reconstructed yaw time history.

A flight proven Adcole digital aspect sensor, similar to the model flown on OAO, is proposed. The sensor will be simplified by using only a single slit per axis and substituting a 7 bit Gray code for the 10 bit binary.

4.2.3.5 Magnetometer

The magnetometer defines the local field vector. From a model of the Earth's field and knowledge of the satellite's position the software program can use the magnetometer output to calculate the vehicle attitude. Magnetometer data will be used during maneuver when earth sensor data is not available. It also serves as backup to either the earth and/or sun sensor in the event of a failure.

A flight proven unit similar to the one flown on P72-1 is proposed. The software is developed.

4.2.3.6 Sequencer

The sequencer furnishes output pulses at preprogrammed quarter orbit times. These outputs will be used to implement the coil torquing schedules generated by the ground software. The sequencer is identical to the model used on S3.

4.2.4 PERFORMANCE

Disturbance torques that must be countered by the attitude control system include the following:

- o Antenna scanning motions
- o Solar radiation torques
- o Station keeping thrust misalignments
- o Orbit regression

Orbit regression is not strictly a disturbance torque, however the bias momentum must be precessed with the inertial rotation of the orbit plane. Orbit regression will average -2.36 degrees per day and can be accommodated by a constant bias of 19600 pole cm. in the pitch electromagnet.

The disturbances are summarized in Table 4.2-4 together with their effect on spacecraft attitude.

Control capability is more than adequate to control expected disturbances. Values are as follows:

Pitch torque:	± 0.052 ft.lb. (0.068 newton meters) available at all times.
Pitch electromagnet: (for spin axis orientation)	6.68 ft.lb.sec/orbit (8.72 newton meter sec./orbit) in quarter orbit switching mode (rotates s/c in declination) 3.45 ft.lb.sec/orbit (4.50 newton meter sec./orbit) in constant coil current mode (rotates s/c in right ascension).
Roll electromagnet: (for wheel momentum desaturation)	8.85 ft.lb.sec/orbit (11.55 newton meter sec/orbit) in half orbit switching mode.

(The electromagnet impulse capabilities are based on 75000 pole cm. Double this capability is available if both coil windings are energized simultaneously).

4.2.5 SOFTWARE

Attitude control and determination is performed using satellite sensor data processed by ground software similar to that developed by Boeing for the STP P72-1 and STP S3 programs. Time correlated telemetry and tracking data will be sent from STDN stations to a data processing center to be used for off-line, non real-time data reduction. Principal outputs are commands for controlling the satellite and an accurate satellite attitude time history.

Information flows are shown in Figure 4.2-5. The raw data received at remote STDN sites consists of attitude and payload telemetry data and tracking information. Payload data is sent directly to the payload data processing center. Attitude data is transmitted to the software program for attitude control and determination. The tracking information is first processed to determine the orbit ephemeris which is then sent to both the software program and the payload data processing center.

TABLE 4.2-4

DISTURBANCE SOURCE	PEAK MOMENTUM OR IMPULSE ROLL FT.LB.SEC	PITCH FT.LB.SEC	YAW FT.LB.SEC	PEAK ERROR ROLL DEG.	PITCH DEG.	YAW DEG.	FREQUENCY HZ
1.0 M Antenna	0.02	0.02	0.04	< 0.01	0	< 0.01	0.33
Solar Radiation	0.07	0.10	0	0.013	0	0	Orbital
Orbit Trim	0	10.5	10.5	2.0	0.5	0	Intermittent
Orbit Regression	0.90 in right ascension			0.06	0	0.06	Cumulative per Orbit

DISTURBANCE IMPULSE SUMMARY

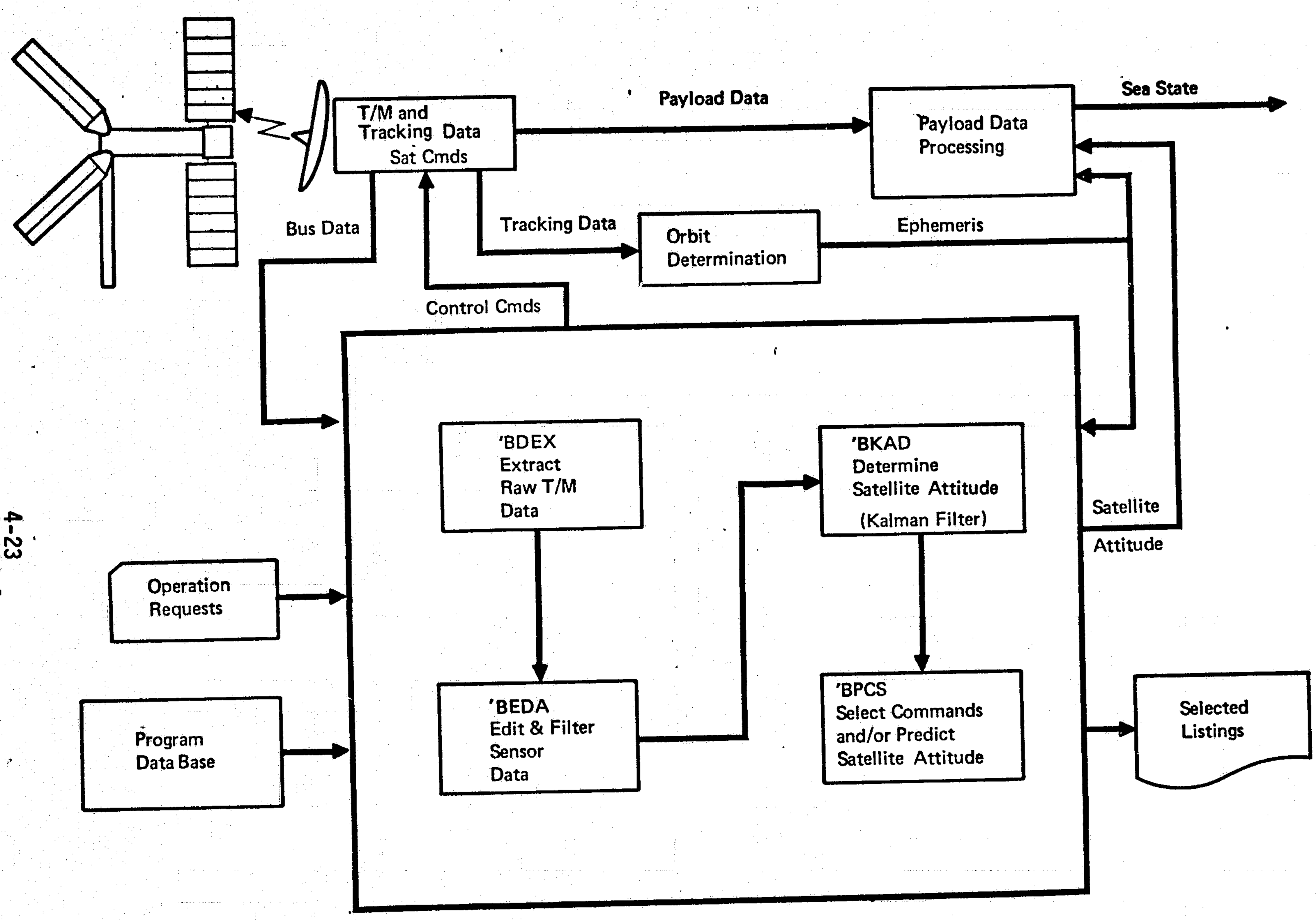

FIGURE 4.2-5: ATTITUDE DETERMINATION AND CONTROL SOFTWARE

4.2.5.1 Attitude Determination

Because the large momentum bias couples roll and yaw in a well defined manner accurate attitude time histories can be reconstructed from two station contacts or a data span approaching one quarter orbit. Accurate attitude can also be determined from real time data transmitted during a single station pass by including sun sensor data.

In operation the software program uses sun sensor or magnetometer data, together with knowledge of the sun line and magnetic field vector, to supplement the horizon sensor information. Corrections are made for earth oblateness, orbit eccentricity, and variations in the effective horizon. Kalman filtering techniques are then employed to obtain a best fit of the sensor data to a time correlated history for the data time span. The attitude data is transmitted to the payload data processing center in the form of coefficients to a polynomial. These operations are performed using the 'BDEX, 'BEDA and 'BKAD subroutines.

4.2.5.2 Attitude Control

The second major function of the software program is to select the sequence of commands required for satellite attitude hold, reorientation maneuvers and wheel desaturation. Subroutine 'BPCS is used for these operations. For the time of interest the program will predict the response of the vehicle to constant and alternating unit coil currents. This data is then used with the existing and desired attitude and wheel speed to define the P_1, P_2 and Q coil levels, Figure 4.2-4. The command sequence is then transmitted to appropriate STDN stations and relayed to the satellite at station pass.

The commands consist of:

Commands	Coil
o Coil current values (2 - 7 bit words) o Coil polarities o High/Low level	Precession Coil
o On/Off o High/Low level o Up/Down o Synchronization drive	Desaturation Coil

During large angle maneuvers frequent contact will be necessary to monitor progress and update commands. Horizon sensor data will be available only intermittently if at all. The magnetometer becomes the primary attitude sensor and while this is less accurate it is more than adequate during maneuver. The magnetometer also provides a backup mode in the event of horizon sensor failure.

The program will have the capability of printing selected outputs such as the command sequence, predicted attitude, attitude determination polynomial coefficients, etc. Outputs are selected by means of operational request cards.

4.2.6 TRADE STUDIES

Control moment gyros, three inertia wheels and a compensated momentum bias system were traded against the uncompensated momentum bias system before the baseline was selected. All are capable of performing the attitude control functions. The baseline was selected primarily for its simplicity and hence implicit reliability and low cost. The system components are listed in Table 4.2-5.

The three CMG and three inertia wheel system are basically similar autonomous concepts. A block diagram is shown in Figure 4.2-6. Pitch and roll data are obtained from the horizon sensor while yaw information is derived by gyrocompassing. The error signals are processed by the drive electronics to accelerate or decelerate the wheels (or torque the CMG's). Desaturation is achieved by the magnetic control logic assembly using wheel speed and magnetometer data.

A compensated bias momentum system, Figure 4.2-7, was also considered. This is similar to the baseline,differing only in the addition of wheel to compensate the momentum of the scanning antennas. The compensating wheel permits the use of a smaller bias momentum wheel. The choice is then between one relatively large wheel (the uncompensated system) and two smaller wheels (one for bias, one for compensation). Simplicity favored the one wheel system. Both concepts can use ground software to close the desaturation and spin axis pointing loops.

One significant advantage of the baseline system is the ability to absorb the disturbances from misaligned station keeping thrusts. All the other candidate systems would require a separate additional subsystem for thrust vector control.

4.2.7 GROWTH

The large bias wheel proposed makes the spacecraft relatively insensitive to antenna scan disturbances. Changes or additions in the antenna complex can thus be accommodated without change to AC&D system design. This includes the imaging radar.

A ground controlled concept has been proposed which leads to a simple on-board system and is fully responsive to present requirements. If requirements should change drastically an autonomous control system might have advantages in providing faster response or more accurate control. Autonomous magnetic control laws are available although none have flown to date, Figure 4.2-8. The Ithaco concept is recommended in which the pitch coil currents are defined by:

$$U = KB_x \phi \text{ for spin axis pointing}$$

and $$U = A \left[\operatorname{sgn}(-\dot{B}_y)\right] \text{ for damping.}$$

Wheel desaturation using the roll coil is relatively straightforward using the tachometer signal.

TABLE 4.2-5

CONCEPT	CMG'S	REACTION WHEELS	MOMENTUM BIAS PLUS ANTENNA SCAN COMPENSATION	MOMENTUM BIAS
	3 CMG'S	3 Wheels	1 Bias Scanwheel	1 Bias Scanwheel
	2 Coils	2 Coils	1 Comp. Wheel	2 Coils
Components	Gyro Unit	Gyro Unit	2 Coils	Magnetometer
	Horizon Sensor	Horizon Sensor	Magnetometer	Sun Sensor
	Magnetometer	Magnetometer	Sun Sensor	FCE
	Sun Sensor	Sun Sensor	FCE	
	FCE	FCE		

APPLICABLE CONTROL CONCEPTS

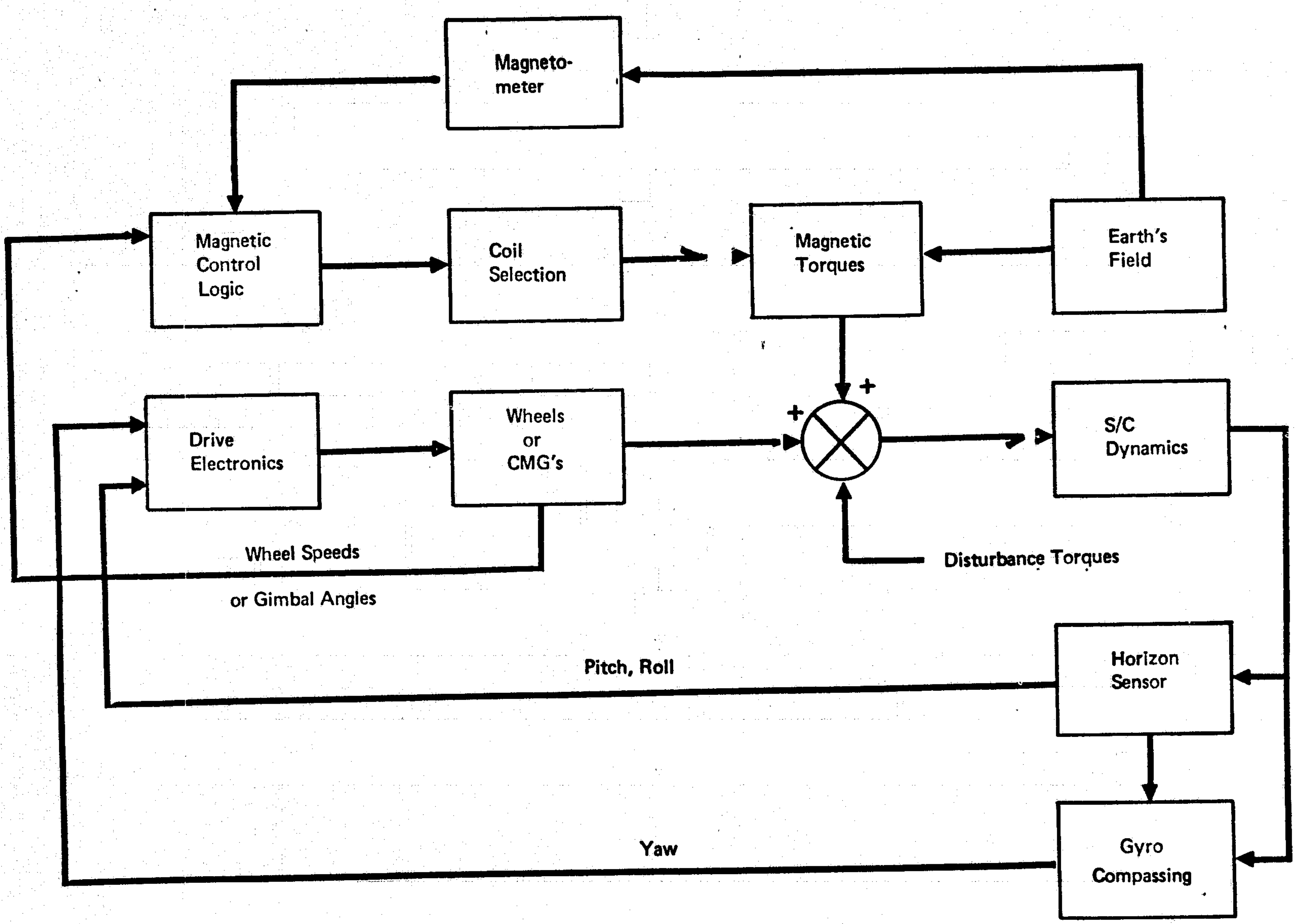

FIGURE 4.2-6: WHEEL OR CMG SYSTEM BLOCK DIAGRAM

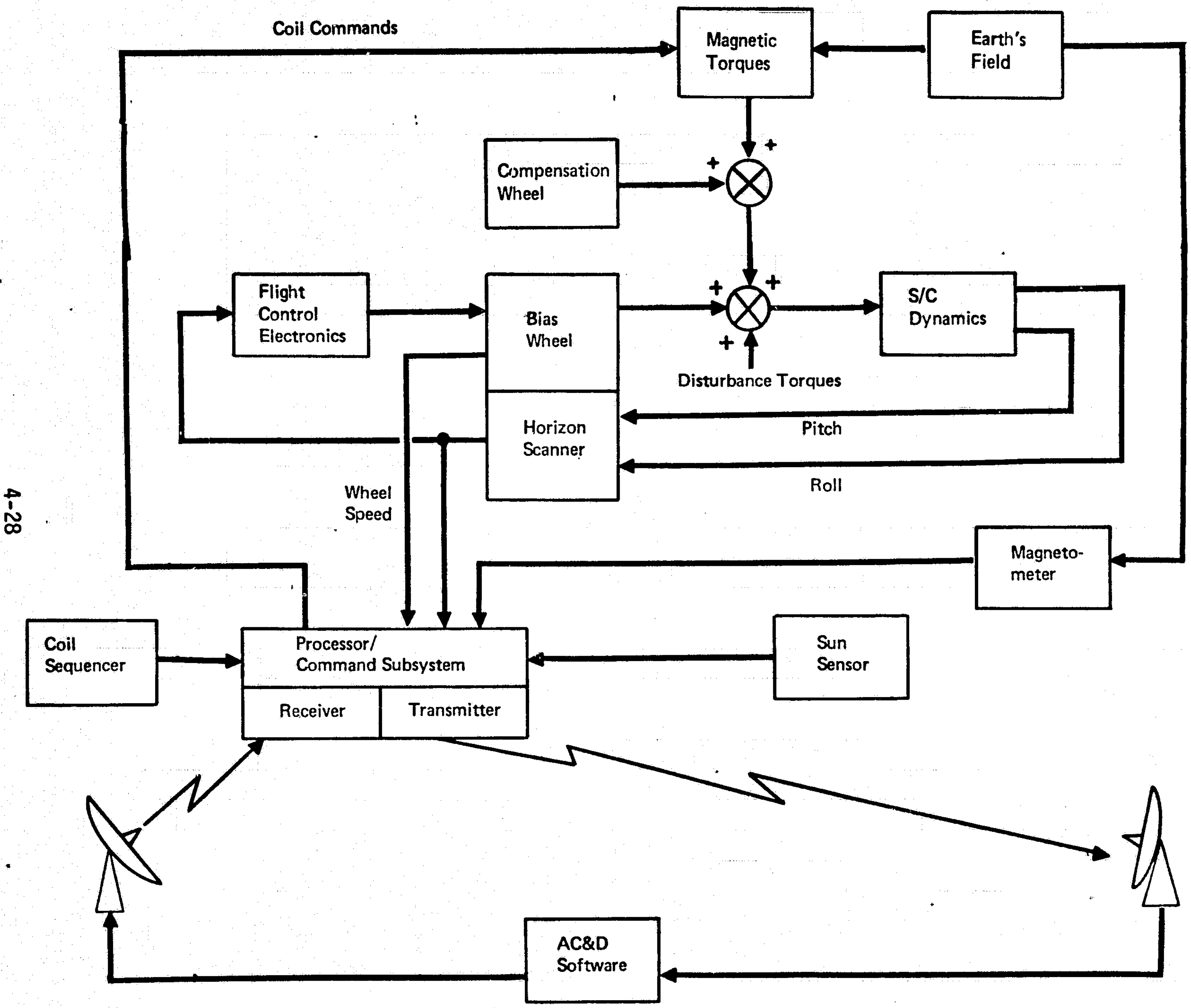

FIGURE 4.2-7: COMPENSATED MOMENTUM BIAS SYSTEM BLOCK DIAGRAM

<u>Wheeler</u>	$U = K\,(B_X\phi - B_Z\psi)$	Attitude Control
(NASA CR-313)	$U = K_1\,B_Z\dot{\phi} - K_2\,B_X\dot{\psi}$	Damping
<u>Shigehara</u>		
(JSR VOL 9 #6)	$U = A\left[\mathrm{sgn}\,(B_X\phi - B_Z\psi)\right]$	Attitude Control
<u>Ithaco</u>	$U = K\,B_X\phi$	Attitude Control
(Itos/Goddard Study)	$U = A\,\mathrm{sgn}\,(-\dot{B}_Y)$	Damping

$$\text{Local field} = \vec{B} = \begin{bmatrix} B_Y \\ B_Y \\ B_Z \end{bmatrix} \qquad \text{Attitude Errors} = \begin{array}{l} \left[\begin{array}{c} \phi \\ 0 \\ \psi \end{array}\right] \begin{array}{l} \text{(roll)} \\ \\ \text{(yaw)} \end{array} \end{array}$$

Pitch Coil Field = U

Figure 4.2-8 <u>AUTONOMOUS MAGNETIC CONTROL CONCEPTS</u>

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Control accuracy better than 0.1 degree 3σ in all axes would require an on-board computer, a star tracker or mapper system and probably three inertia wheels such as OAO. A general purpose computer, such as the CDC 469, could be used to provide injection guidance, on orbit control and desaturation and some or all of the attitude determination functions.

All attitude acquisition capability could be provided by adding additional reaction jets (fast response) or autonomous magnetic control (low response) and a coarse sun sensor.

An alternative method of accomplishing the 180 degree yaw turn maneuver is to add a single additional hydrazine thruster that produces a roll torque. A 0.5 lb. motor on a 4 ft. arm can complete the maneuver in less than 8 minutes with full pitch control being maintained throughout. A relatively short period of an orbit or two may be required for damping and to bring the spin axis to within ± 0.5 degree of its new orientation. Relieving the magnetic system of the reorientation task would permit reduction of the electromagnets from 150,000 pole cm. to 50,000 pole cm.

4.3 TELEMETRY, TRACKING AND COMMAND

4.3.1 Introduction. The design of a Telemetry, Tracking, and Command (TT&C) System is described in this part of the report. This design achieves all specified and assumed performance requirements for the SEASAT-A Spacecraft (S/C).

4.3.1.1 TT&C Requirements and Design Approach. The TT&C System performance requirements (see Table 4.3-1) have been established based on the data handling, command execution, and tracking criteria defined in documentation supplied by JPL. Because no firm baseline set of criteria was established, certain assumptions have had to be made; therefore, the final detail design must await resolution of payload interfaces.

A design approach has been used where a nominal set of system parameters for achieving the TT&C functions is established and then the equipment functional requirements to accomplish these functions have been defined. This approach is a systematic design procedure which clearly identified the tradeoffs among the alternate designs and, as such, provides a basis for selection of a design which can be achieved in the most efficient and economical manner.

The TT&C system has been configured utilizing existing equipment that has been space qualified on previous programs and will require minimum modification to meet the SEASAT-A requirements. A limited industry survey was conducted to identify available space-qualified equipment to meet the requirements of the SEASAT-A program. The information contained in this section provides the best system configuration based on this limited survey. The possible use of a general purpose computer processor (e.g. CDC-469 or Autonetics D200 series) was considered for command decoding and programming. Due to the need for additional interfacing circuit design and new software, this more expensive approach was not pursued in favor of existing hardware for a dedicated system. If, in the final design analysis, a requirement for a more universal TT&C is established, the use of such a computer for command and telemetry processing should be considered.

Another TT&C system design approach that was considered was the possibility of using existing Integrated Command and Telemetry "party line" (Data Bus) type systems. The result of this investigation concluded that the present SEASAT-A TT&C system performance requirements could not justify the cost of the increased flexibility and capacity that these type systems could provide. If future payload data requirements indicate that selective sampling and special command requirements are required, the addressable Telemetry and Command type systems should be reevaluated.

Two interface definitions need to be established prior to design implementation of the payload module and the bus module. One interface is the signal characteristics between payloads and the Telemetry and Command subsystems. The other is the configuration of the STDN during the time period that the S/C will be flown needs to be well defined in order to assure compatibility between the S/C TT&C system and the ground equipment.

TABLE 4.3-1

TT&C SYSTEM FUNCTIONAL REQUIREMENTS

The TT&C System shall:

1. Be compatible with the NASA Spaceflight Tracking and Data Network (STDN).

2. Provide a telemetry subsystem that will sample, encode, and format all S/C data for either immediate transmission to the STDN ground station, or for storage and subsequent transmission to the STDN.

3. Provide a command subsystem capable of initiating S/C operations either by real-time commands or by programmable, stored commands.

4. Generate a timing reference for use, if required, to satisfy S/C timing requirements.

5. Provide a RF subsystem which will receive command and tracking data from the STDN and transmit telemetry and tracking data to the STDN.

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4.3.1.2 TT&C System Configuration. The proposed TT&C system for the SEASAT-A spacecraft is shown in the simplified block diagram of Figure 4.3-1. The Telemetry Subsystem incorporates redundant PCM processors and tape recorders to meet all requirements for encoding, formating, record/playback, and baseband modulation of all payload module and S/C bus data. The Command Subsystem consists of a selectively redundant Command Decoder/ Programmer and Clock assembly and a Relay Box for real-time or automatic stored program (up to 18 hours) execution of all ground commands to the payload module and S/C bus subsystems. The RF Subsystem (RFS) incorporates redundant coherent transponders and associated switches, diplexers, hybrids, etc., and an S-band antenna to receive and transmit signals for communications with the STDN.

The TT&C design incorporates dual-redundancy of all active components and makes maximum use of flight-proven and space qualified off-the-shelf equipment. The redundant elements (except command receiving and decoding circuits) are on standby and require no power until activated by command. TT&C equipment status, performance and characteristics are summarized in Table 4.3-2.

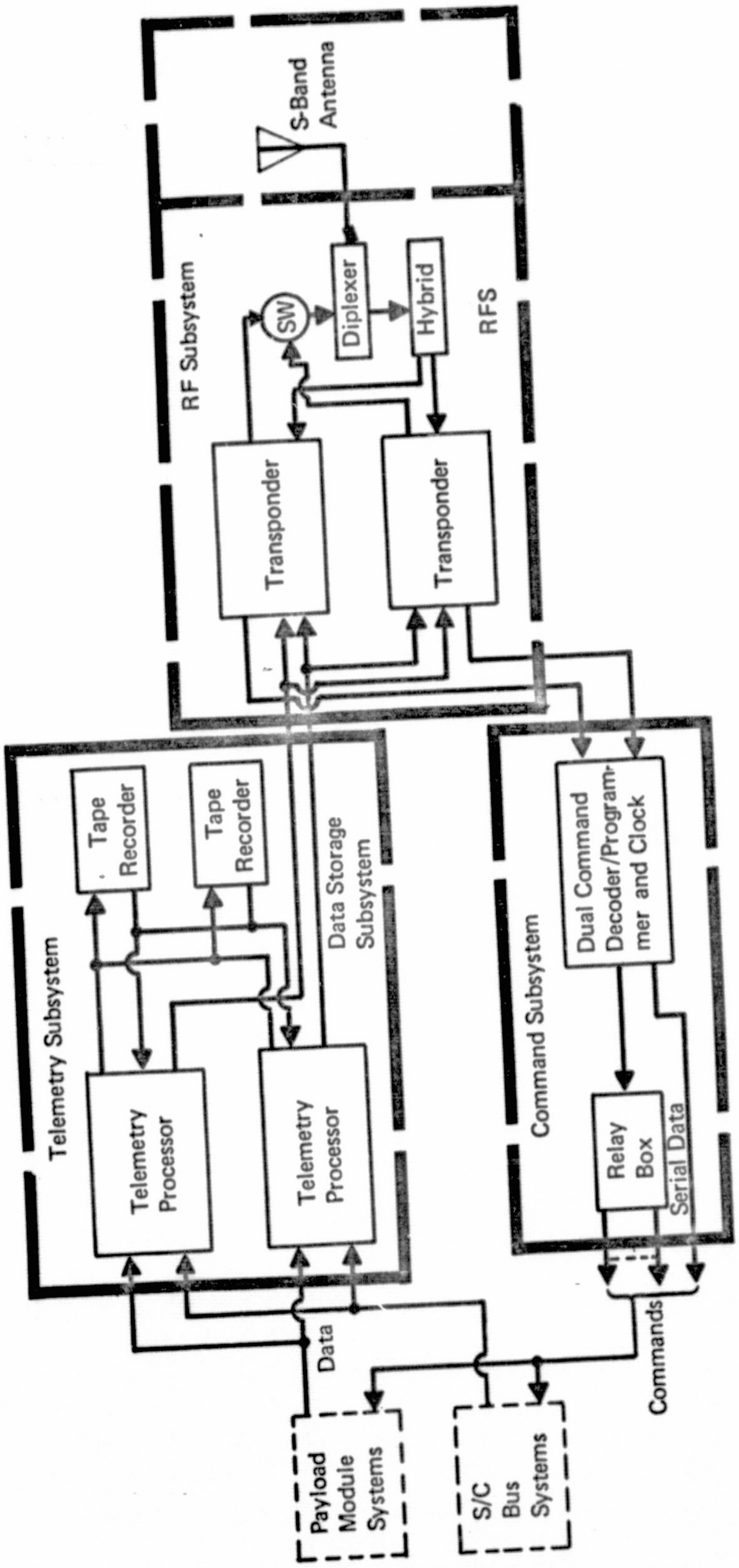

Figure 4.3-1: TT&C System Block Diagram

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4.3.2 Telemetry Subsystem

4.3.2.1 Telemetry Processing. Payload and Engineering data processing is accomplished with the processor/baseband unit, Figure 4.3-2, whose design features include:

- o Space proven design.
- o Combined Processor and Baseband Sections.
- o Internal three point calibration.
- o Output bit rate synchronized to spacecraft clock.
- o Dual redundancy via ground commands.
- o Selectable data formats.

The unit will accept analog and digital data from the payloads and spacecraft bus subsystems, combine all signals into a selected format, and route the T/M signal to the RF System for transmission to ground facilities. The unit quantizes all analog data into 8-bit words, providing a 3-sigma accuracy of ± 0.4%.

Data Formating - The payload data and engineering data are accepted through the analog and digital multiplexers of the processor. The analog data is converted into digital words by the A/D converter and then sequentially combined with the digital data into one composite serial data format. In addition, this format contains frame sync words, sub-com identification, time code words and command verification. The output bit rate is the sum total of the requirements for all the payloads and the S/C bus. This is estimated to be 36 Kbps, including approximately 10% spares.

Various modes at different data rates from the payload module were indicated as a baseline requirement. Up to four different, ground commandable, formats are capable of being programmed in the telemetry processor. Unless a large difference in bit rates are required between the various modes from the payload, one data format would probably satisfy all requirements. The three modes established for this study could be handled by a single data format.

In order to ensure adequate bit transition density for ease of bit synchronization, care will be taken in designing the formats such that no large blocks of data with all bits "1" or all bits "0" exists in the data format. This is especially important when one of the NRZ codes are used.

Signal Routing - The Processor provides all signal routing functions: For real time transmission of data, the 36 Kbps format is continuously routed from the processor to the low rate SCO in the baseband section, and after modulation the signal is routed to the transmitter. For data storage, the 36 Kbps format is routed to both recorders and storage is accomplished by commanding the appropriate recorder. Stored data (720 Kbps) is routed to the high rate SCO in the baseband on playback of

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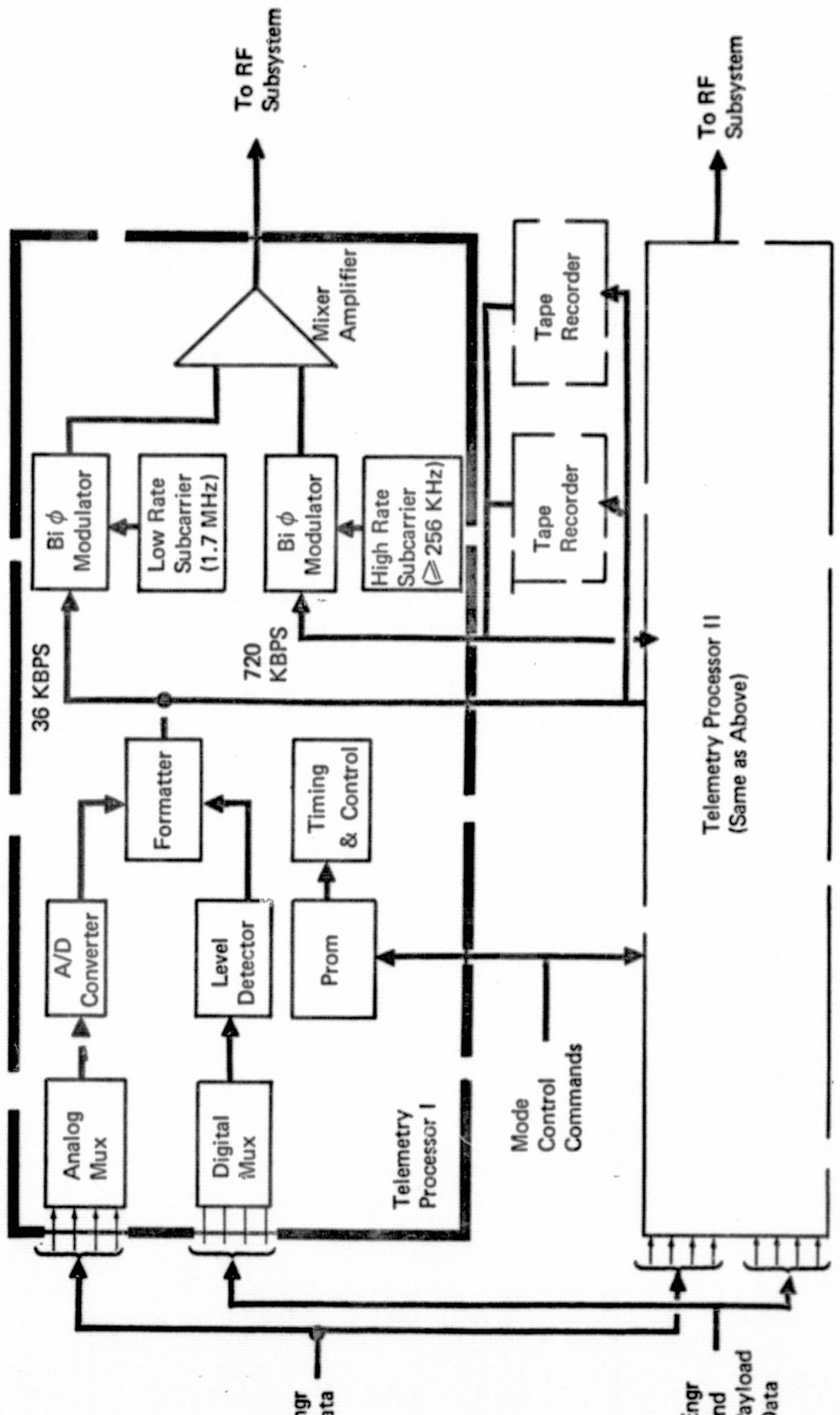

Figure 4.3-2: Telemetry & Data Storage Subsystems Block Diagram

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TABLE 4.3-2 TT&C EQUIPMENT CHARACTERISTICS

PROCESSOR (TELEDYNE DS-714) - 2 REQ.

STATUS	SPACE QUALIFIED (S-3)
ANALOG INPUTS	192
DIGITAL INPUTS	120
SAMPLE RATES	UP TO 128 sps
WORD LENGTH	8 BITS
OUTPUT FORMATS	36 KBPS NRZ-L PCM
SIZE (IN)/VOL (IN^3)	8 X 4.5 X 4.3/155
WT/DC POWER	6 LB/13 W

RECORDER (ODETIC DDS-3100) - 2 REQ.

STATUS	SPACE QUALIFIED (~~72-2~~ STP P72-1)
NO. OF TRACKS	5
RECORDING TECHNIQUE	DIRECT DOUBLE DENSITY
TOTAL CAPACITY, EACH	1.536 X 10^9 BITS
TAPE SIZE	1,800 FT 1-MIL ¼ IN.
BIT PACKING DENSITY	14.2 KBPI MAX.
SIGNAL INPUT	NRZ-L PCM
RECORD/PLAYBACK RATE	36/720 KBPS
RECORD/PLAYBACK SPEED	2.54/50.8 IPS
SIGNAL OUTPUT	NRZ-L PCM
OVERALL BIT ERROR RATE	10^6 MAX.
SIZE (IN.)/VOL ($IN.^3$)	12 X 9 X 6.5 /710
WT/DC POWER	16.8 LB/20-30W

COMMAND DECODER/PROGRAMMER AND CLOCK (CALCOMP-NIMBUS E/F TYPE)- 1 REQ.

STATUS	SPACE QUALIFIED (ERTS)
INPUT	50 BIT WORD, STDN COMPATIBLE
OUTPUT	480 REAL TIME, 30 STORED
SIZE (IN.)/VOL ($IN.^3$)	6 X 8 X 13/624
WT/DC POWER	22/26 W

RELAY BOX -(BOEING 10-70218) - 1 REQ.

STATUS	SPACE QUALIFIED (S-3) MODIFIED
INPUT	114 PULSES MAX
OUTPUT	80 LATCH, 42 PULSE
SIZE (IN.)/VOL ($IN.^3$)	1.3 X 4 X 2.3/119
WT/DC POWER	4 LB/NA

TABLE 4.3-2: TT&C EQUIPMENT CHARACTERISTICS (continued)

RFS (MOTOROLA) - 1 REQ.	"M" SERIES	ERTS
STATUS	TO BE QUALIFIED BY 1975	QUALIFIED BY THE ERTS PROGRAM 1971
SIZE (IN.)/VOL (IN.^{3})	3.0 x 8.0 x 16.0/384	8.0 x 6.0 x 13/624
WT/DC POWER	16.0 LBS/13.0 WATTS	24 LBS/28 WATTS
RECEIVER (2 REQ)		
FREQUENCY	2025 TO 2120 MHz	2020 TO 2120 MHz
FREQUENCY STABILITY	20 PARTS PER 10^6	20 PARTS PER 10^6
NOISE FIGURE	7.5 dB	8.0 dB
IF BANDWIDTH (3 dB)	1000 Hz	800 Hz
SENSITIVITY	-130 dBm	-127 dBm
IMAGE REJECTION	$>$80 dB	$>$80 dB
MOD FORMAT (CMD)	70 KHz SUBCARRIER	70 KHz SUBCARRIER
DATA RATE (CMD)	1000 BPS W/8-BIT SUB-BIT DECODER	1000 BPS W/8-BIT SUB-BIT DECODER
TRANSMITTER (2 REQ.)		
FREQUENCY	2200 TO 2300 MHz	2200 TO 2300 MHz
BANDWIDTH (3 dB)	4 MHz	$>$5 MHz
RF POWER	$<$2 WATTS	$<$2 WATTS
MODULATION	PM	PM
MODULATION SENSITIVITY	0.6 RAD/VOLT	1.0 RAD/VOLT

DIPLEXER - HYBRID COUPLER (MOTOROLA ERTS)

STATUS	SPACE QUALIFIED (ERTS)
TRANSMIT	
LOSS	1.0 dB
ISOLATION	75 dB
BANDWIDTH	43 MHz
FREQUENCY	2287 MHz
RECEIVE (EACH CHANNEL)	
LOSS	4.3 dB
ISOLATION	86 dB
BANDWIDTH	56 MHz
FREQUENCY	2106 MHz

TABLE 4.3-2: TT&C EQUIPMENT CHARACTERISTICS (Continued)

S-BAND ANTENNA (BOEING 233-10043-1) - 1 REQ.

STATUS	SPACE QUALIFIED (STP P72-1 AND S3)
TYPE	LOG SPIRAL
GAIN	+0.6 dBi 160.4^{o} COVERAGE (2250 MHz)
FREQUENCY	1750 TO 2300 MHz
VSWR	1.5:1 MAX.
POLARIZATION	RIGHT HAND CIRCULAR
WT/DC POWER	0.7 LB/NA

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the recorder. After modulation, the signal is routed to the transmitter. Simultaneous transmission of both real time data and stored data is possible. The low rate subcarrier (256 KHz) is also generated whenever the baseband is powered up. The high rate subcarrier (1.7 MHz) is controlled on and off via ground command.

Data Timing - The Processor contains an internal oscillator, slaved to the main spacecraft clock in the command sub-system. All processor timing functions are derived from this source. This includes timing the format generation and shift and control pulses for serial transfer of digital data from the payloads to the processor for formating into the telemetry format. The time code word from the command sub-system is read into the Telemetry format each main frame and is synchronized to the main frame leading edge. Synchronization of the Telemetry format to the central clock allows resolution of data in the format to an accuracy of one bit (27.8 s).

4.3.2.2 Data Storage. Telemetry storage is provided by two Odetics DDS-3100 tape recorders connected to operate interchangeably. Two recorders are required for continuous recording, i.e., for simultaneous recording and playback of telemetry data. Each recorder provides 1.54 X 10^9 bits of storage and can record up to 11.8 hours of 36 Kbps NRZ-L telemetry data at maximum bit density (14.2 Kbps). However, since 11.8 hours are not required, a lower bit density is recommended. Control of the recorders is via real-time command with the capability to initiate "RECORD," "PLAYBACK," FAST FORWARD," "STOP" and power ON/OFF for each recorder. Telemetry data is provided on recorder mode, track location, etc., to permit on-orbit status determination, failure detection, and analysis. The recorders employ a serial, multiple-track operation (five tracks), utilizing 1800 feet of tape. Data loss due to track switching amounts to less than five seconds per track. One orbit of data could be stores on less than one track of the tape; therefore, no data would be lost due to track switching. The random bit-error-rate is 1 part in 10^6, including block errors.

Operationally, one recorder could be used for data acquisition with the second recorder providing playback of previously recorded data during station contacts. Upon completion of the playback mode (up to 3 station contacts) the recorder functions will be interchanged. This allows data to be recorded continuously.

4.3.2.3 Telemetry Equipment

Processor/Baseband Unit. The selected Processor/Baseband Unit is a Teledyne Telemetry Company Model DS-714, which is a modified version of their model DS-704 PCM System. The Processor section contains essentially the same circuits as the DS-704, arranged to meet the sampling rates and other requirements of SEASAT-A. The circuit elements are high-reliability, low power, TTL-integrated circuits for the logic circuits and high reliability MOS-FET gates for analog and digital data input switching. The baseband section is a mix of low-power TTL integrated circuit elements and discrete components.

The Telemetry industry is now developing PCM telemetry equipment using CMOS logic instead of TTL logic. CMOS has the advantage of consuming only a small fraction of the power required by TTL logic, providing simplified design, and

increasing reliability. CMOS is presently being used on the S3 program in many of the experiment packages. These packages have been Space Qualified. Whenever SEASAT is committed to hardware, telemetry equipment using CMOS logic should be readily available and will be seriously considered.

Tape Recorder - The proposed tape recorder is an Odetics Model DDS-3100 currently being qualified for space flights on the STP 72-2 satellite. This recorder is a derivative of the DDS-3000 originally used on SESP 70-1 satellite, with later versions used on SESP 71-2 and 72-1 satellites.

The predicted bit error rate, including block errors, is equal to or better than 10^{-6}. Track switching occurs in five seconds or less for a total loss of 720 kilobits in recording five tracks of data. Operating life is expected to be 6000 to 8000 hours of actual operation based on tape/head, motor, and negator spring life tests conducted by Odetics. Two Odetics recorders on the STP P72-1 satellite have each accumulated over 2600 hours of record/playback time and data users report bit error rates remain less than 1×10^{-6}.

At present, Goddard Space Flight Center (GSFC), has a specification out to the industry (specification 73-15033) for a Standard Satellite Tape Recorder with a capacity of 1×10^{9} bits. If this recorder is space qualified prior to SEASAT being committed to hardware, it will probably be the appropriate unit to use.

A second alternate recorder to consider is the Odetics DDS-5000. The primary difference between this recorder and the DDS-3100 is that the DDS-5000 has no track switching and therefore no data interrupts. It is expected that the DDS-5000 will be space qualified by the end of 1974. It is being developed for use on a classified Air Force satellite.

4.3.3 Command Subsystem. The command subsystem receives the 125 bit per second bit stream from the baseband demodulator and sub-bit decoder in the RF subsystem. The Command Decoder/Programmer and Clock assembly decodes and verifies correctness of commands, provides memory for delayed command, generates a primary time reference for the S/C, and generates and routes command signals via a control relay box to the payloads and S/C bus module subsystems.

4.3.3.1 Command Decoding. The command decoding equipment is an existing design which has successfully flown on Nimbus and ERTS satellites. The decoder consists of two identical decoder sections which receive the command signals from the redundant sub-bit decoders located in the RF subsystem. The Decoder/Programmer has the capability of executing 480 real time commands formated in 50-bit words and storing up to 30 commands for execution from memory. Execution of stored commands occurs at any desired integer second after initiation of a "Go" timer. This memory can also be programmed for repeated execution of any of the 30 commands in storage. The repeated execution times are entered as the time to repeat after the first execution.

To achieve reliability, the Command Decoder/Programmer and Clock is designed to be selectively redundant. Selective redundancy refers to switching various sections of the total redundant sections, to obtain a full operating subsystem. Figure 4.3-3 shows a functional block diagram of the full redundant subsystem.

4.3.3.2 Command Modes. The command subsystem can operate in four modes: (1) real time command, (2) command data storage, (3) time code set, and (4) serial data output. The real time command mode decodes a 50-bit message and initiates the command through the decoding matrix and relay drive circuits. The command storage data mode transfers input data to the command storage section for processing and storage. The command storage section performs data storage by using dynamic type MOS shift registers which are capable of holding 50 bits of information. There are two command storage sections which provide the capability of 30 stored commands, 15 in each section. The command storage is not considered to be redundant as both can be active at the same time. However, redundancy can be achieved by storing the same commands with the same time of execution in both command storage sections. The third operating mode of the command decoder is to transfer a time code set message to the time code generation section. The time code generation section performs two functions - it provides real time coded signal and generates the precision frequencies for the spacecraft subsystems. The time code generation section divides the signal from the 3.2 MHz oscillator to generate the various coherent precision frequencies for other subsystems. Signal division is accomplished with flip-flop ripple counters. The last mode of operation is the serial data transfer mode. As the serial data is transmitted to the S/C, the command decoder receives, checks the address, partity and key bits. These items are stripped out and only 36 data bits are sent to the S/C subsystem.

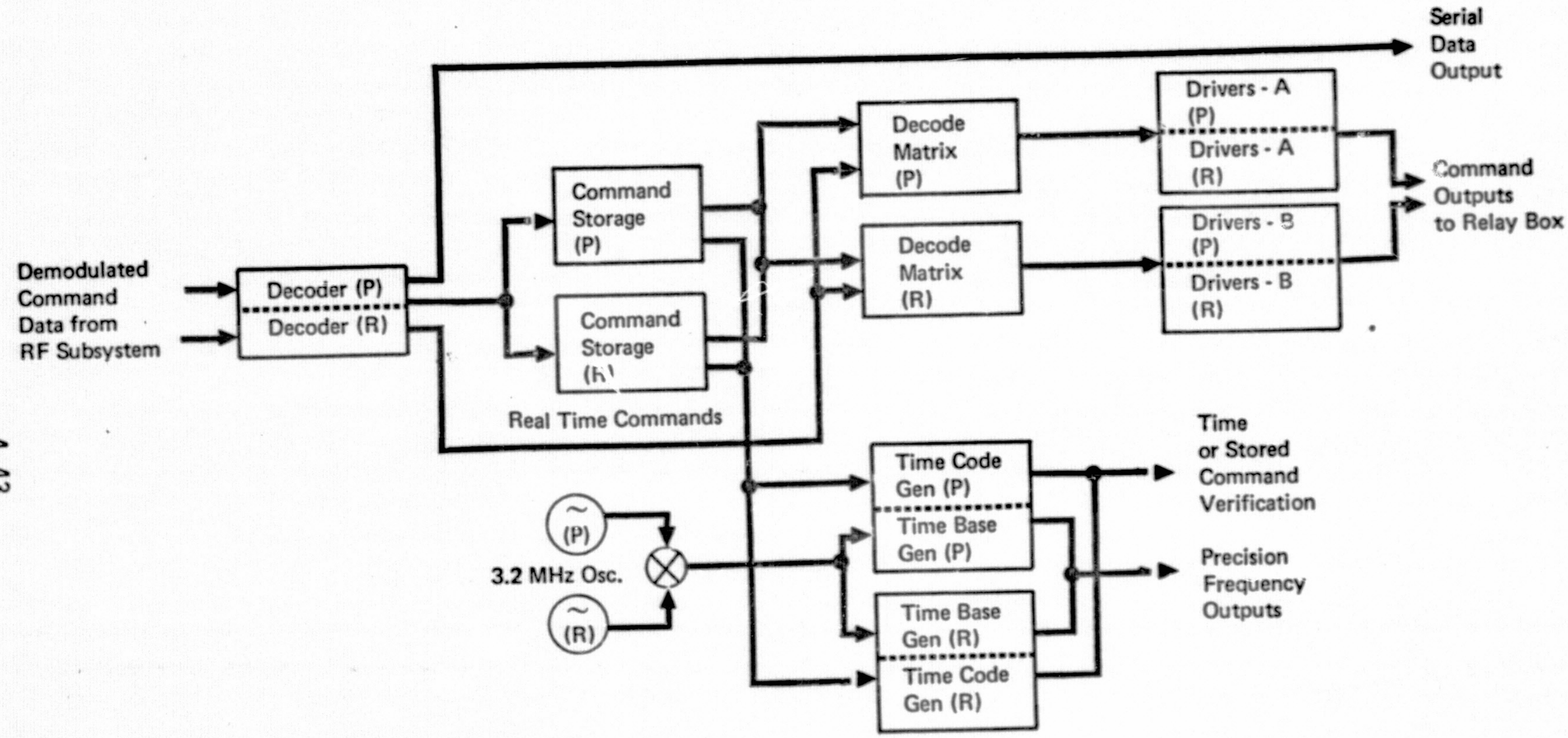

Figure 4.3-3: Command Decoder/Programmer and Clock Block Diagram

4.3.3.3 Command Equipment. The physical and electrical characteristics of the Command subsystem equipment is summarized in Table 4.3-2.

Command Decoder/Programmer and Clock. The Command Decoder/Programmer and Clock is an existing off-the-shelf unit built by California Computer Products Inc.; Anaheim, California. This flight-proven equipment incorporates high reliability, integrated circuit design, and has been used on both the NIMBUS and ERTS satellite programs.

Relay Box. The relay box is a self-contained, Boeing-built unit with relays, Printed Circuit (PC) boards, and packaging identical to that used in the S-3 satellite relay box. The relays consist of latching and non-latching types and are mounted on the PC boards along with other required discrete components. Heavier relays are mounted directly to the assembly housing.

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4.3.4 R.F. Subsystem

4.3.4.1 Uplink. Uplink reception is accomplished via the S-Band antenna, the R.F. Subsystem (RFS) Diplexer, Hybrid Coupler, and Receivers as shown in Figure 4.3-4. The uplink signal at 2090-2120 MHz (this frequency band should be extended to 2025-2120 MHz by the time SEASAT-A flies) is split equally by the hybrid to feed dual redundant receivers, thereby assuring the required reliability for uplink command signals.

The S-Band phase-lock receiver is part of the STDN Compatible Unified-S-Band Transponder set. It phase demodulates the uplink carrier and separates out the command 70 KHz frequency modulated (FM) subcarrier. The 70 KHz discriminator demodulates this subcarrier to produce a 2 KHz sine wave which is phase-shift-keyed by a 1 kilobit per second PCM command. A 1 KHz sine wave which is added to the 2 KHz phase-shift-keyed signal is used as a clock for the command decoder. This is the standard command modulation used by STDN. A sub-bit decoder within each receiver decodes the 1000 bps command data previously coded into 8 symbols per information bit by the STDN and outputs 125 bps NRZ command data, bit strobe, and a message duration pulse to the Command Subsystem dual redundant decoders.

The receivers separate out the sequential tone ranging signals and provide them to an associated Transmitter for modulation on the downlink S-Band carrier. Each receiver also supplies a coherent drive to an associated Transmitter to phase-lock the downlink to the uplink carrier. Modulation indices of 0.8_r(Command 70 KHz subcarrier) and 0.6_r(Sequential tone ranging) supports the link analysis shown in Table 4.3-4.

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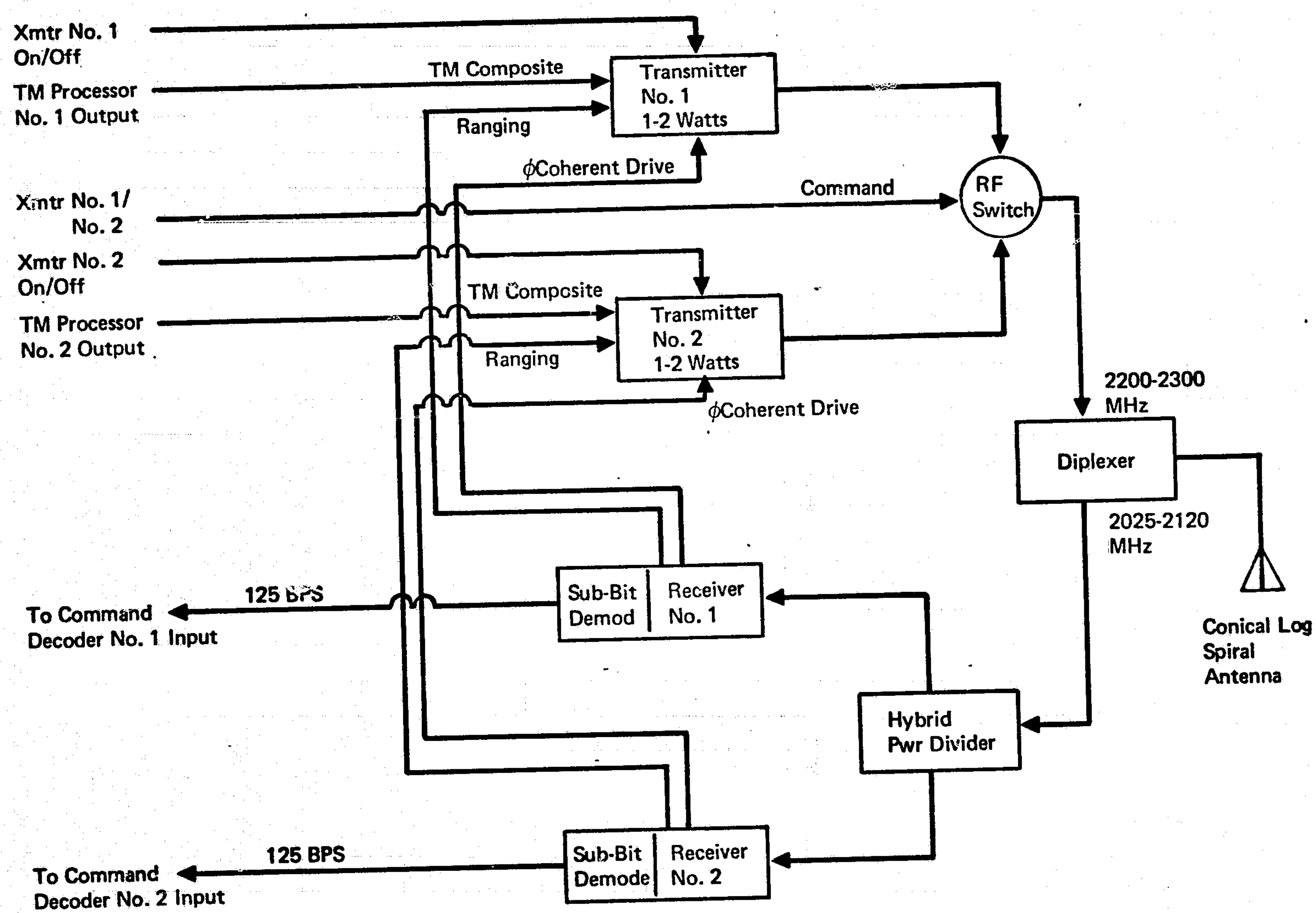

Figure 4.3-4: TT&C Radio Frequency Subsystem Block Diagram

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4.3.4.2 Downlink. Downlink transmission at 2200-2300 MHz contains telemetry and the turn around ranging and is phase coherent with the uplink S-Band. Dual redundant Transmitters are selectable by ground command as shown in Figure 4.3-4. The downlink power is isolated from the receiver by a diplexer which feeds the signal to the S-Band antenna.

The voltage-controlled oscillator (VCO) of the selected Transmitter is phase locked to the respective receiver coherent drive signal. An auxiliary oscillator is provided in each transmitter in the event that coherent drive is lost for any reason (e.g., no up-link signal). The selected Transmitter is phase modulated by a composite signal from the selected telemetry processor and a sequential tone ranging signal consisting of 500 KHz, 100 KHz, and lower frequency resolution tones. This total baseband signal is downlinked with modulation indices of 0.65_r,(low rate subcarrier), 1.5_r (high rate subcarrier), and 0.3_r (sequential tone ranging). Sequential tone ranging will be STDN operational by 1975 (per NASA Goddard) and is included herein in lieu of the present PRN ranging technique. The ranging turn-around ratio is set to 2:1 Uplink to Downlink. A transmitter output power of 1 watt supports the link analysis of Table 4.3-3. These parameters must be reevaluated after a better definition of the STDN is obtained.

4.3.4.3 TT&C Link Analysis. Downlink and uplink analyses are shown in Table 4.3-3 and Table 4.3-4, respectively. STDN compatible values (as currently understood) were used for all applicable parameters. Both the uplink and downlink have a $\geq$ 6 dB performance margin at a downlink transmitter power of .87 watts. This is recommended to be increased to 1 or 2 watts and thereby provide an excess of 6 dB margin on the downlink to allow for adverse tolerance buildup in various parameters. A 0.6 dB atmospheric loss is allocated for S-Band transmission at low ground station elevation angles. A reduction of the uplinked power (1 Kw to 100 watts, or less, ould reduce possible overloading of the satellite receiver during minimum range passes.

4.3.4.4 Antenna Performance. Figure 4.3-5 depicts a representative antenna pattern for the Boeing conical log spiral antenna used in previous programs. This antenna is identical to that on Boeing's S3 Satellite and similar to earlier versions flown on SESP 70-1 and SESP P72-1. It features a broadband balum and balanced twin-lead feed system to provide a VSWR of less than 1.5:1 over the 1750-2300 MHz band previously used. The measured RHCP gain over about $\pm 60^{\circ}$ coverage is greater than 0 dB, per Table 4.3-2.

TABLE 4.3-4 RF LINK ANALYSIS SUMMARY - UPLINK

NO.	RF LINK PARAMETER	VALUE	SOURCE
1.	TOTAL TRANSMITTER (1 Kw)	+ 63.0 dBm	STDN Parameter
2.	TRANSMITTING CIRCUIT LOSS	---	Included in 3
3.	TRANSMITTING ANTENNA GAIN (30' DISH)	+ 43.0 dB	STDN Parameter
4.	TRANSMITTING ANT. POINTING LOSS	---	Included in 3
5.	SPACE LOSS (SR-1724 NM)	-169.0 dB	Calculated @ 5^{0} EL
6.	POLARIZATION LOSS	---	RHCP
7.	ATMOSPHERIC LOSS	- 0.6 dB	Worst Case
8.	RECEIVER POINTING LOSS	0.0 dB	Stabilized
9.	RECEIVING ANTENNA GAIN	0.0 dB	Estimated
10.	RECEIVING CIRCUIT LOSSES	- 2.4 dB	Estimated
11.	TOTAL RECEIVED POWER	- 66.0 dBm	1 through 10
12.	RECEIVER NOISE DENSITY (290^{o}C)	-166.5 dBm/Hz	@ N.F. = 7.5 dB
	CARRIER ACQUISITION PERFORMANCE		
13.	ACQUISITION THRESHOLD POWER	-130.0 dBm	Specified
14.	PERFORMANCE MARGIN	+ 64.0 dB	11 less 13
	COMMAND PERFORMANCE		
15.	MODULATION LOSS	- 6.4 dB	MI = .8
16.	COMMAND CH. NOISE B.W. (4 KHz)	- 36.0 dB	Estimate
17.	REQUIRED SNR IN NOISE B.W.	+ 20.0 dB	Estimate (10^{-6} BER)
18.	COMMAND THRESHOLD POWER	-108.1 dBm	12 + 15 + 16 + 17
19.	PERFORMANCE MARGIN	+ 38.1 dB	11 less 18
	SEQUENTIAL TONE RANGING PERFORMANCE		
20.	MODULATION LOSS	- 8.8 dB	MI = .6
21.	RANGING CH. NOISE B.W. (4 KHz)?	- 36.0 dB	Estimate
22.	REQUIRED SNR IN NOISE B.W.	+ 20.0 dB	Estimate (10^{-6} BER)
23.	RANGING THRESHOLD POWER	-105.7 dB	12 + 20 + 21 + 22
24.	PERFORMANCE MARGIN	+ 34.7 dB	11 less 23

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TABLE 4.3-3 RF LINK ANALYSIS SUMMARY - DOWNLINK

NO.	RF LINK PARAMETER	VALUE	SOURCE
1.	TOTAL TRANSMITTER POWER (.871W)	29.40 dBm	10 minus 2 thru 9
2.	S/C CIRCUIT LOSS	2.0 dB	Estimated
3.	TRANSMITTING ANT. GAIN	0 dB	Estimated
4.	TRANSMITTING ANT. POINT LOSS	---	Included in 3
5.	SPACE LOSS (SR = 1724 NM)	-169.75 dB	Calculated @ 5° Elevation
6.	POLARIZATION LOSS	0	RHCP
7.	ATMOSPHERIC LOSS	-0.6 dB	Worst case
8.	RECEIVING ANTENNA GAIN (30' DISH)	+44 dB	STDN Parameter
9.	RECEIVING CIRCUIT LOSS	---	Included in 8
10.	TOTAL RECEIVED POWER	-98.96 dBm	25 - 24
11.	RECEIVER NOISE DENSITY	-176.3 dBm/Hz	Calculated
	CARRIER PHASE LOCK ACQUISITION PERFORMANCE		
12.	MODULATION LOSS	-7.51 dB	Calculated
13.	RECEIVED CARRIER POWER	-106.47 dBm	10 + 12
14.	CARRIER NOISE BW (600 Hz)	27.78 dB Hz	Calculated
15.	REQUIRED SNR IN NOISE BW	7 dB	STDN Parameter
16.	THRESHOLD POWER	-141.52 dBm	11 + 14 + 15
17.	PERFORMANCE MARGIN	35.05 dB	13 - 16
	PCM TELEMETRY DATA (LOW RATE SCO) PERFORMANCE		
18.	MODULATION LOSS	-13.78 dB	Calculated (MI = .65 Rad.)
19.	RECEIVED SC POWER	-112.74 dBm	10 + 18
20.	SC NOISE BW (36 KBPS)	45.56 dB Hz	Calculated
21.	REQUIRED SNR IN NOISE BW	12.0 dB	Estimated
22.	THRESHOLD POWER	-118.74 dBm	11 + 20 + 21
23.	PERFORMANCE MARGIN	6.0 dB	19 - 22
	PCM TELEMETRY DATA (HIGH RATE SCO) PERFORMANCE		
24.	MODULATION LOSS	-3.75 dB	Calculated (MI = 1.5 Rad.)
25.	RECEIVED SC POWER	-102.71 dBm	23 + 29
26.	SC NOISE BW (360 KBPS)	55.56 dB Hz	Calculated
27.	REQUIRED SNR IN NOISE BW	12.0 dB	Estimated
28.	THRESHOLD POWER	-108.74 dBm	11 + 26 + 27
29.	PERFORMANCE MARGIN	6.03 dB	25 - 23
	RANGING PERFORMANCE		
30.	MODULATION LOSS	-17.70 dB	Calculated (MI = .3 Rad.)
31.	RECEIVED POWER	-116.66 dBm	10 + 30
32.	NOISE BW (1 Hz)	0 dB Hz	STDN Parameter
33.	REQUIRED SNR IN NOISE BW	35 dB/Hz	STDN Parameter
34.	THRESHOLD POWER	-141.3 dBm	11 + 32 + 33
35.	PERFORMANCE MARGIN	24.64 dB	31 - 34

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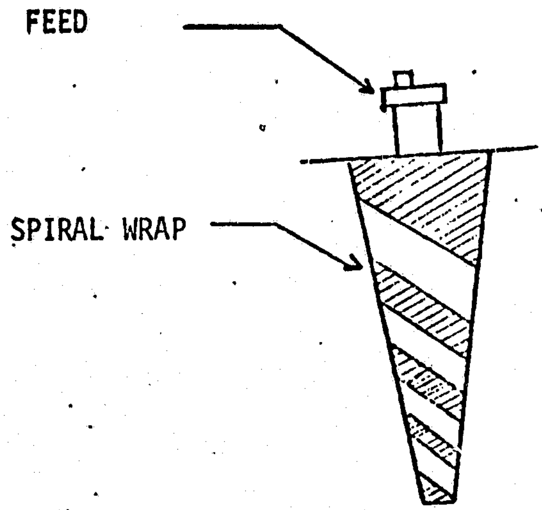

CONICAL LOG SPIRAL ANTENNA (S-BAND)

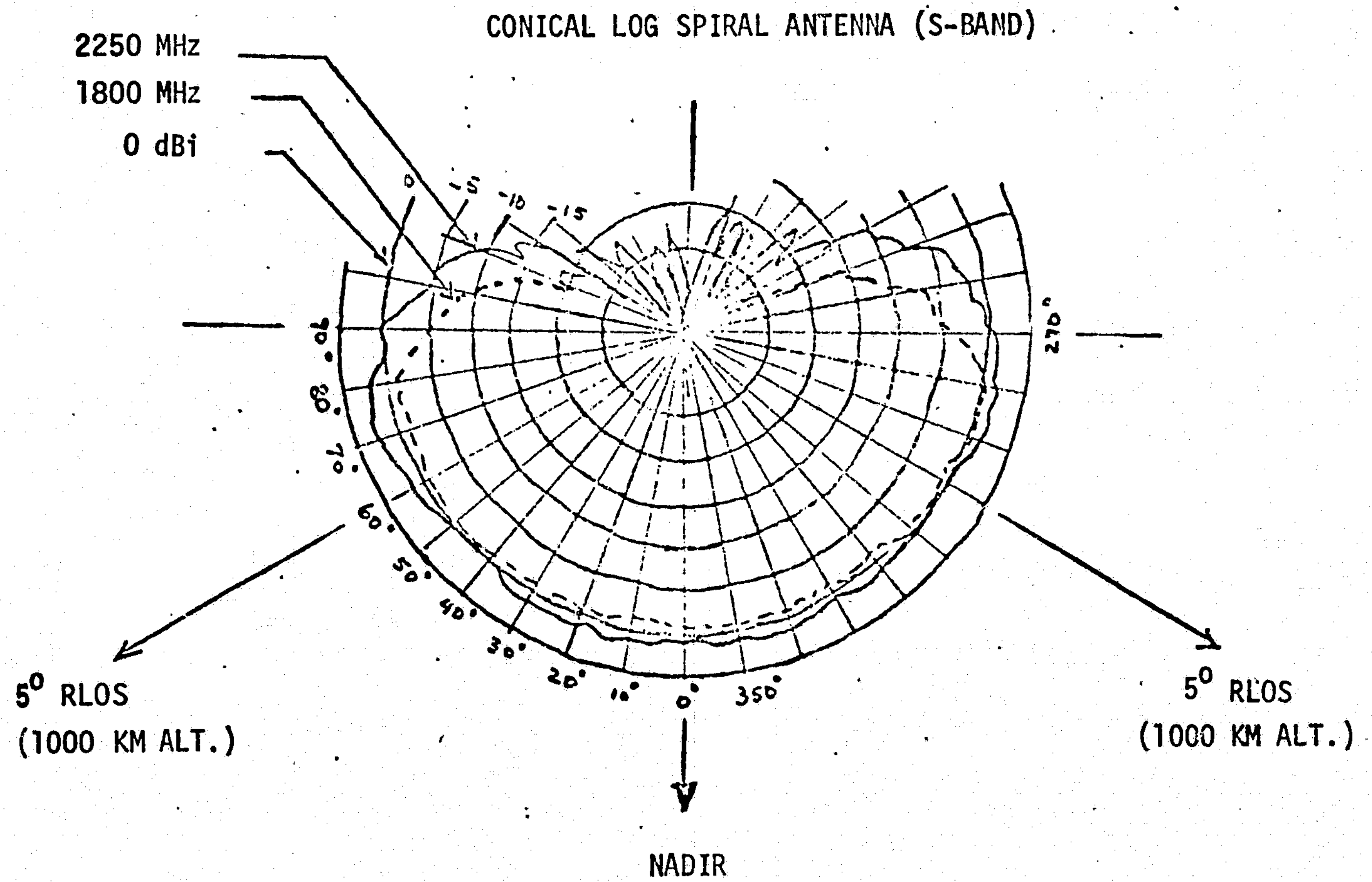

FIGURE 4.3-5: TT&C ANTENNA PERFORMANCE

4.3.4.5 Hardware Implementation. The physical and electrical characteristics of the RF subsystem equipment is summarized in Table 4.3-2.

RFS - The Motorola M-Series transponder is proposed for use in the SEASAT-A RF subsystem. This transponder is being developed and qualified for space application and has demonstrated an improved performance and reliability; reduced size, weight and power consumption; and lower cost than older transponders such as Motorola's ERTS units. These improvements have been achieved through the use of current circuit and packaging technology and a family of special purpose space-qualified monolithic integrated circuits. A special sub-carrier (70 KHz) demodulator and sub-bit decoder (8 to 1) will be added to the present M-series configurations. This additional module will use previously developed circuitry and technology from other programs. Additional testing required for qualification will be minimal.

Motorola's ERTS Unified S-Band transponder was also considered, however this model uses older technology and would need upgrading in some areas such as the power amplifier, etc. The M-series is expected to be more reliable and somewhat less expensive.

RF Components - The RF components including the switch, diplexer, and hybrid power divider will be standard space qualified equipment. The R.F. Switch is a switchable ferrite circulator from the ERTS program. It is of the same power level capability and has an established flight history. The diplexer and hybrid is a combination unit used on other programs such as VO-75 or ERTS. The antenna, a Boeing-build unit, is space qualified and is presently being used on the Boeing S3 satellite.

Special Test Equipment - Some special ground test equipment for test and checkout will be required. Special test support equipment for ERTS is in the NASA inventory and could be used for SEASAT in the unmodified form (for Motorola ERTS Transponder) or with minimal modification for the Motorola M-Series transponder as proposed.

TT&C Schedule - Figure 4.3-5 shows the schedule and major milestones for the design, fabrication and test of the major TT&C elements.

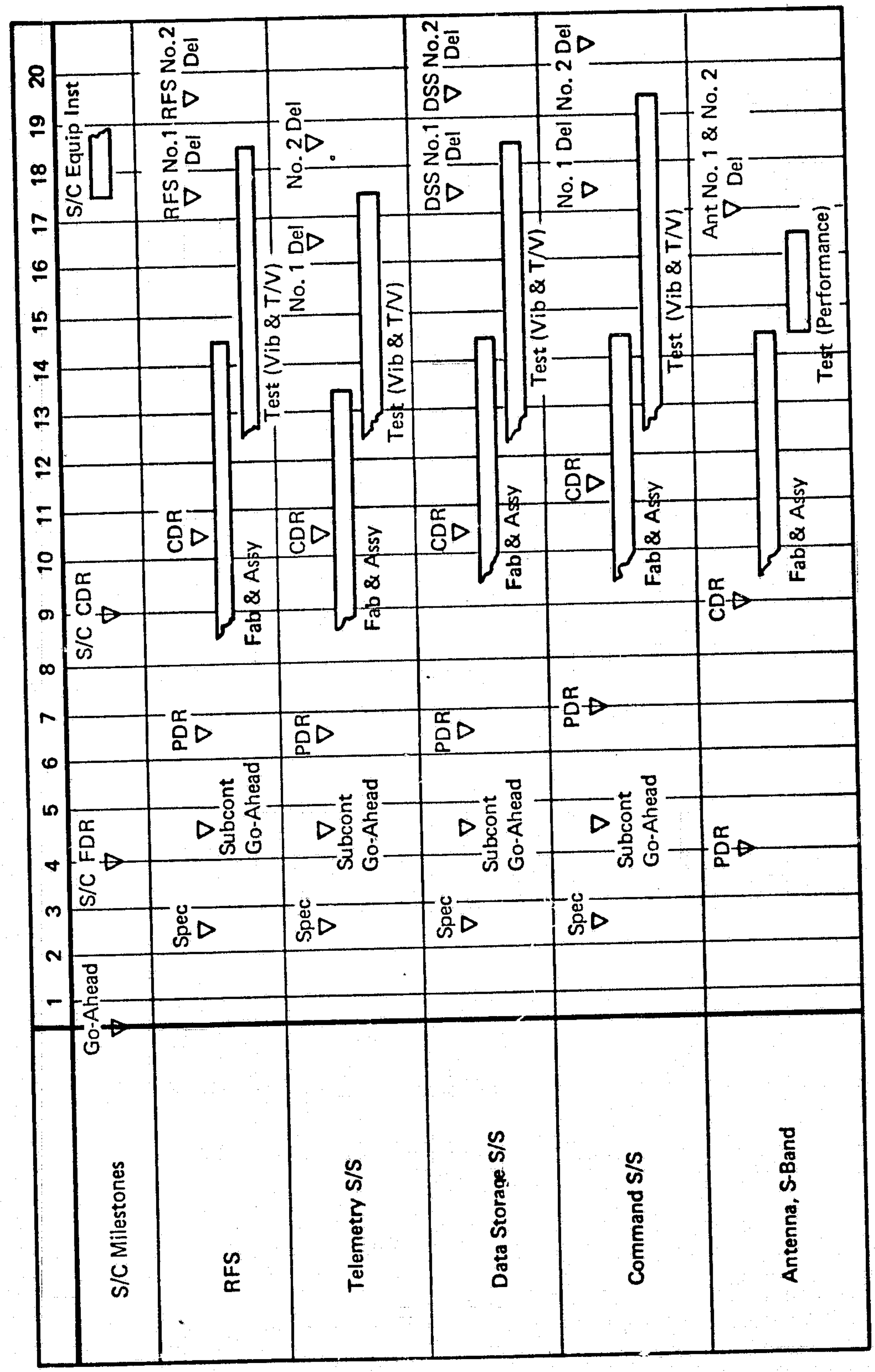

Figure 4.3-5: TT&C Schedule

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4.4 ELECTRICAL POWER

The baseline Electrical Power and Distribution Subsystem, shown functionally on Figure 4.4-1, uses 110 square feet of single axis articulated solar array, four 18 AH nickle-cadmium batteries, four battery charge controllers, four boost regulators, eight shunt regulators, a power relay box and a basic two wire distribution system with a single point ground, to provide a simple, reliable and cost effective system.

The power subsystem design is based on the following requirements:

- o Electrical load - 450 watts average 28 ± 4 volts
- o Mission duration - one year
- o Orbit altitude - 700 to 1000 km
- o Beta angles - 0 to ± 90°
- o Maximum occult durations - 35.3 minutes
- o Number of charge/discharge cycles - 4320
- o Vehicle rotated 180° as required
- o Maximum use of qualified hardware
- o Minimum cost

The solar array is comprised of a 55 square foot panel on each side of the spacecraft bus, interconnected with a single axis slipring drive assembly. Mission time lines and orbit Beta angles shown on Figure 4.4-2 together with power drain requirements for on-board spacecraft and payload sensor equipment, shown in Table 4.4-1, were used to size the power system.

TABLE 4.4-1

SEASAT ELECTRICAL LOADS

SENSORS	
ALTIMETER	125 W
ALTIMETER PREPROCESSOR	10 W
MICROWAVE RADIOMETERS	50 W
SCATTEROMETER	150 W
DOPPLER BEACON	6 W
VISIBLE & IR RADIOMETERS	8 W
R F SWITCH	3 W
TOTALS	352 W
MAXIMUM ORBITAL AVERAGE	350 W
SPACECRAFT SUBSYSTEMS AVERAGE	100 W
TOTAL AVERAGE ORBITAL POWER	450 W

SEASAT POWER SUBSYSTEM - BASELINE

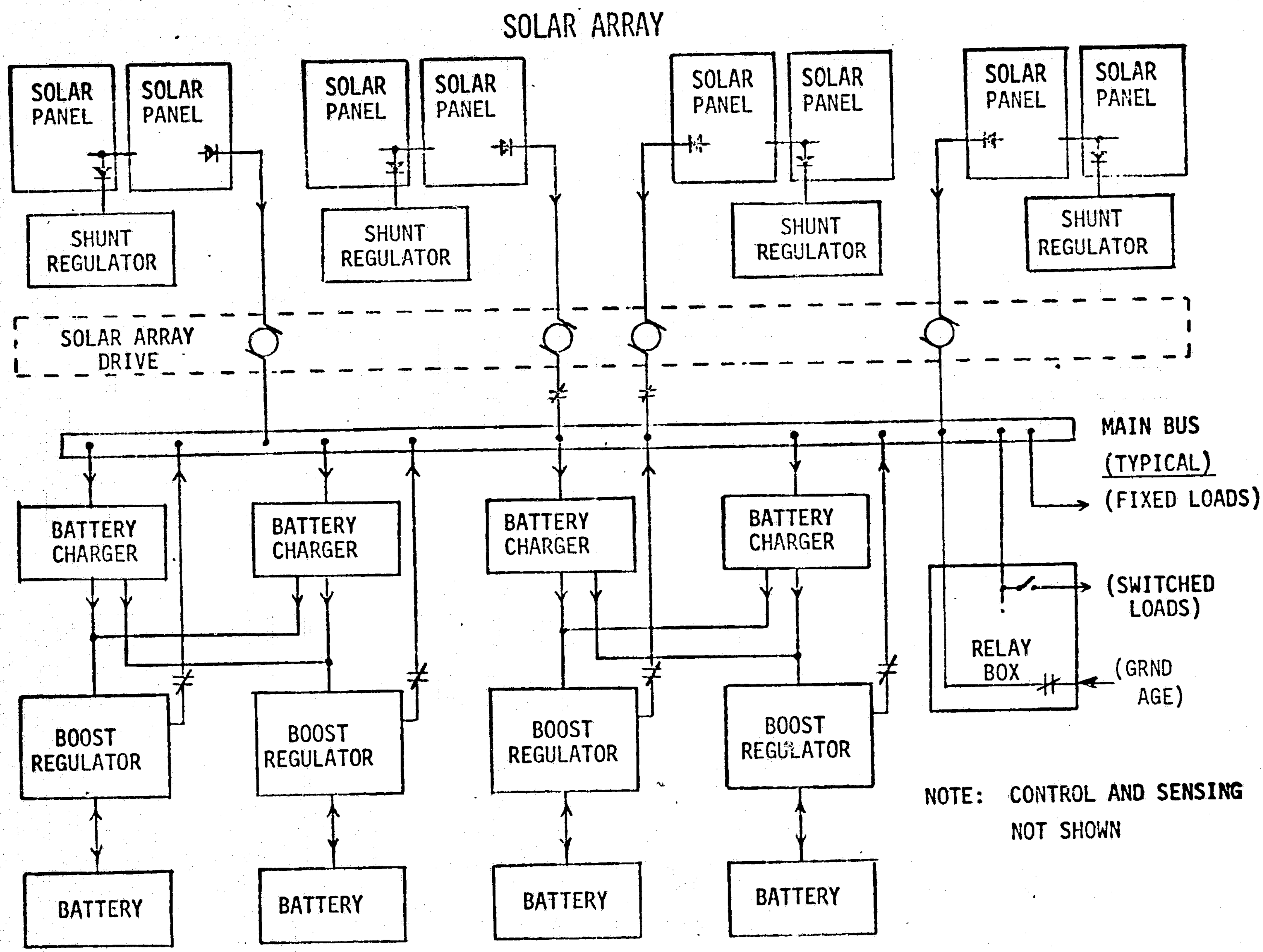
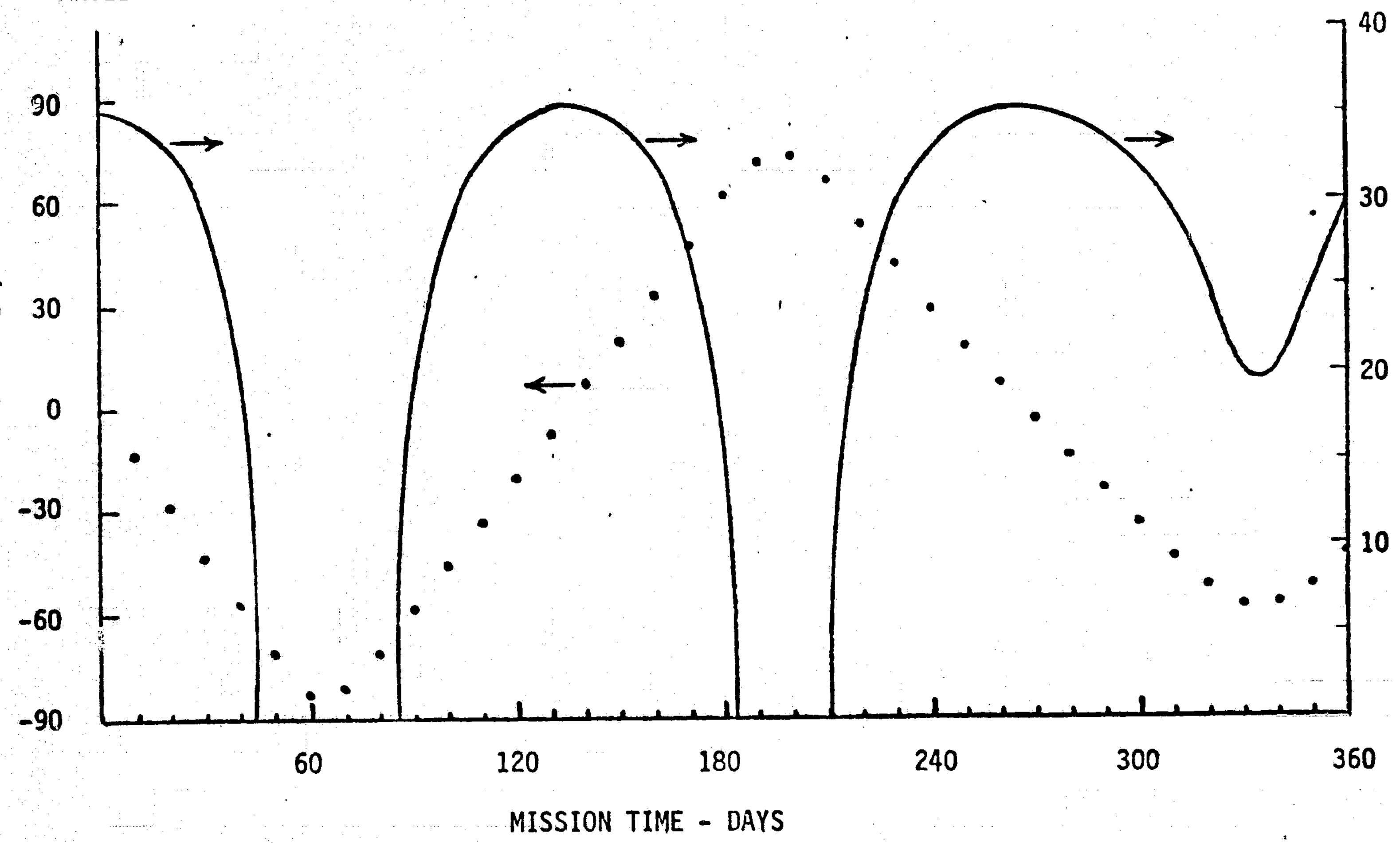


FIGURE 4.4-1

TYPICAL MISSION - ORBIT BETA ANGLES/OCCULT TIMES



- 700 KM ALTITUDE
- 110 DEGREE INCLINATION
- 21 MAR 78 LAUNCH

FIGURE 4.4-2

THE BOEING COMPANY
SEATTLE, WASHINGTON

4.4 ELECTRICAL POWER (Continued)

Energy storage to support spacecraft occultation periods is provided by four nickle cadmium 18-cell batteries, each having a capacity of 15 ampere hours. Control for battery charging is provided by four modified and redundant charge control units previously used on P72-1 satellite and interfacing with the same batteries.

Voltage control for the spacecraft bus is provided by eight redundant shunt regulators, installed on the solar panel support structure, to dissipate excess energy produced by the array during lower power demand periods of the mission. During occultation energy is drawn from the spacecraft battery system through redundant boost regulators that provide minimum bus voltage of 26.25 ± 1.0 VDC.

The power system relay box provides capability for controlling battery charge rate, transfer from ground power to on-board power, isolation of individual batteries, charge controllers, boost or shunt regulators if failed.

Power distribution for the spacecraft employs a multiple channel, two wire, power and ground return, connected to a 28^{+4}_{-2} VDC main bus and ground return bus having a single point ground located at the power system relay box.

4.4.1 Power Subsystem Design. Using the orbit time histories in Figure 4.4-2 and the electrical loads shown in Table 4.4-1, the power required for selected mission times is derived. A number of orbit Beta angles and occultation periods are shown in Table 4.4-2. From this data it can be seen that an array size of 905 watts average power is required for the lowest Beta angles and longest occultation period if we are to assure battery recharge while carrying the full spacecraft load of 450 watts. Boost regulator losses during their operating period (occult) and battery energy conversion efficiency is accounted for during the effective time periods. It is noted that with higher Beta angles and shorter occultation periods less total energy per orbit is required. The design of the solar array thus must accommodate the lowest Beta case and is optimized for maximum power for that condition. Shunt regulators must be sized to dissipate excess energy; however, efficient thermal control, shunt regulator sizing, and lower cost solar array orientation systems result when the array output is optimized along the average power required profile as shown in Figure 4.4-3.

In considering hardware for the required functions, power system hardware from three Boeing programs were reviewed:

- o MVM'73
- o P72-1
- o STP - S3

This review compared system complexity, component multiples for capacity or performance, ease of providing redundancy and system growth potential. Factors and relative merit are summarized in the following Table, 4.4-3.

TABLE 4.4-2.

POWER REQUIREMENTS VS BETA ANGLE

β ANGLE/OCCULT MIN*	$0^{\circ}/35.3$	$20^{\circ}/34.2$	$30^{\circ}/33.0$	$50^{\circ}/26.2$	$60^{\circ}/17.0$	$64.5\text{-}90^{\circ}/0$
BOOST REGULATOR						
PWR, OUTPUT	450 W	450 W	450 W	450 W	450 W	450 W
PWR INPUT	510 W	510 W	510 W	510 W	510 W	510 W
BATTERIES & CHARGER						
DISCH AMPS	25	25	25	25	25	
DISCH AH	14.7	14.3	13.8	10.9	7.1	
CHG AH	16.9	16.5	15.9	12.5	8.16	
CHG AVG AMPS	15.7	15.3	14.7	11.6	7.55	
CHG AVG WATTS	455	443	426	336	218	
CHG TRICKLE C/40						43.5 W
SOLAR ARRAY						
LOAD WATTS	450	450	450	450	450	450
BATT CHG	455	443	426	336	218	43.5
TOTAL	905	893	876	786	668	493.5

* BASED ON 700 KM ORBIT ALTITUDE

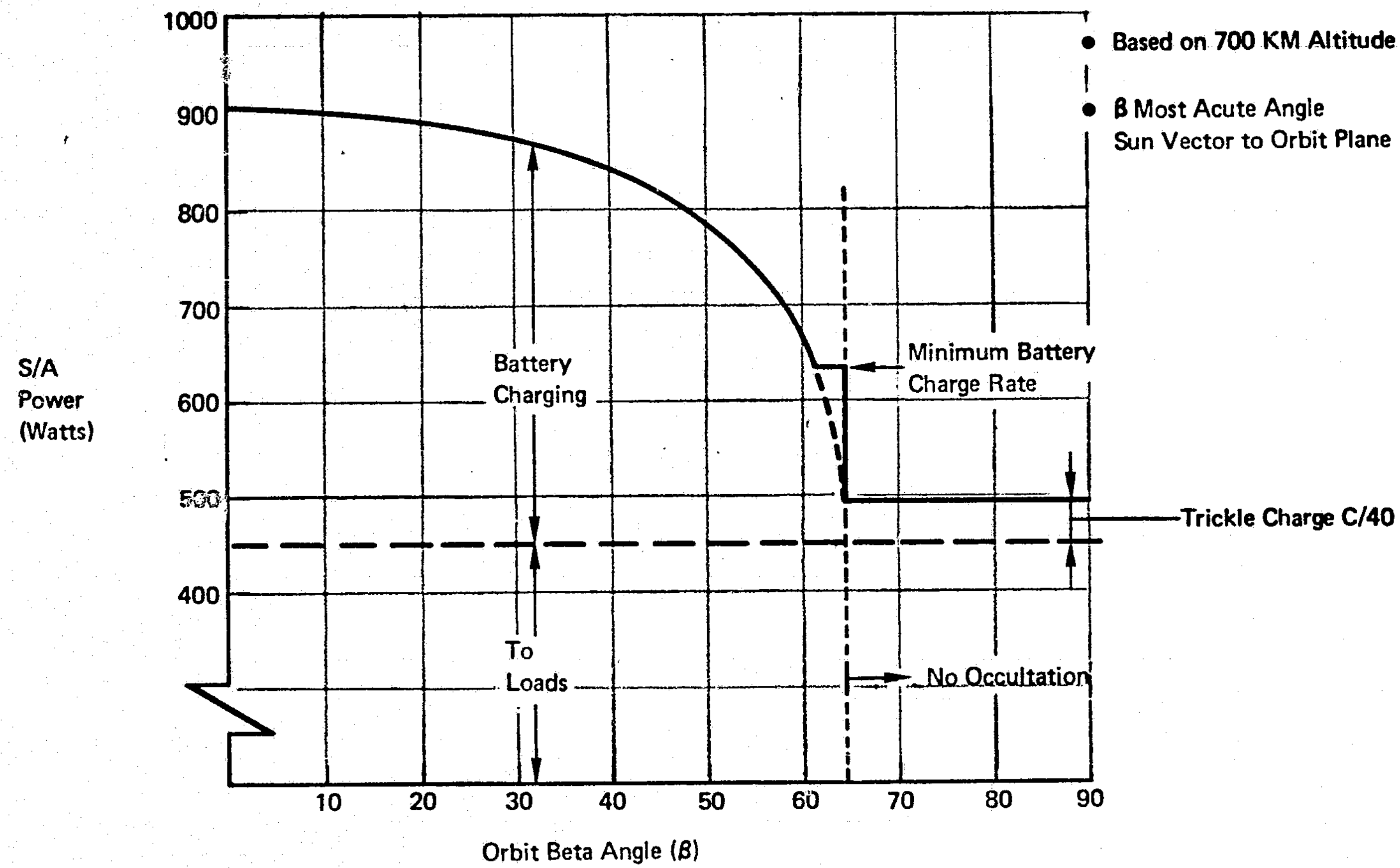


Figure 4.4-3: Solar Array Power Requirements

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TABLE 4.4-3

MAJOR CONSIDERATIONS

CANDIDATE	ADVANTAGES	DISADVANTAGES
S3	o SIMPLE o LOW COST o HIGH EFFICIENCY	o LOAD GROWTH-PARALLEL BATTERIES - DEDICATED S/A SECTIONS REQUIRED - DIODE ISOLATION REQUIRED - NO FORCED LOAD SHARING o LARGE OUTPUT VOLTAGE VARIATIONS
P72-1	o MEDIUM COMPLEXITY o MEDIUM COST o MEDIUM EFFICIENCY o FORCED BATT LOAD SHARING o GOOD DC VOLTAGE REG o LOAD GROWTH (MODULAR CONCEPT)	o THERMAL INTERFACE NOT OPTIMUM o PACKAGING NOT OPTIMUM
MVM'73	o EXCELLENT AC VOLTAGE REG o MINIMIZES POWER SUPPLIES IN AC UTILIZATION EQUIP	o HIGHEST COMPLEXITY o HIGHEST COST o VERY LARGE DC VOLTAGE VARIATIONS o NO FORCED LOAD SHARING FOR REDUNDANT COMPONENTS o LOAD GROWTH (LIMITED)

Factors summarized in the above table assisted in selection of proven hardware components for the SEASAT study. Power subsystem components and variations required are listed in Table 4.4-4. Changes to P72-1 hardware required for the SEASAT mission are summarized in Table 4.4-5. Physical parameters and required quantities of the power subsystem hardware components are shown in Table 4.4-6.

TABLE 1.4-4

BASELINE ELECTRICAL POWER SUBSYSTEM COMPONENTS

DESCRIPTION	QUANTITY	TYPICAL SUPPLIERS	STATUS	REMARKS
SOLAR PANELS	(110 SQ. FT. TOTAL AREA)	SPECTROLAB EOS	MODIFIED DESIGN	SIMILAR TO P95
SHUNT REGULATORS	8	HUGHES ENG MAG	MODIFIED DESIGN	QUALIFIED DESIGN CONCEPT
BOOST REGULATORS	4	HUGHES	P72-1	MINOR CHANGE TO PROVIDE ADDITIONAL LOAD SHARING CAPABILITY
BATTERY CHARGERS	4	ENG MAG	P72-1	MINOR CHANGE TO PROVIDE TRICKLE CHARGE CAPABILITY
BATTERIES	4 (THREE 6-CELL PACKS/BATTERY)	HUGHES	P72-1	BASELINE
RELAY BOX	1	BOEING	MODIFIED DESIGN	SAME DESIGN CONCEPT FOR P72-1 AND S3 PROGRAMS
SOLAR ARRAY DRIVE	1	TRW BALL BROS.		SAME OR SIMILAR TO NIMBUS DRIVE

RO 74-100

3-20-74

C-2

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TABLE 4.4-5

CHANGES TO P72-1 COMPONENTS

	72-1	SEASAT	REASON FOR CHANGE
BATTERY CHARGER			
CONSTANT CURRENT	3.0/6.0 AMPS	3.5/4.7 AMPS	POWER REQUIREMENTS FOR SEASAT MISSION
TRICKLE CHARGE	NONE	C/40 RATE (GROUND COMMAND)	TO PREVENT EXCESSIVE OVERCHARGE/ MINIMIZE TEMPERATURE DURING NO-OCCULT ORBITS
BOOST REGULATOR			
LOAD SHARING	2 PARALLEL UNITS	4 PARALLEL UNITS	TO EQUALIZE DOD OF ALL FOUR BATTERIES

TABLE 4.4-6

POWER SUBSYSTEM COMPONENTS

COMPONENT	NUMBER UNITS	SIZE/UNIT	WEIGHT/UNIT
SOLAR ARRAY	1	110 SQ. FT.	120 LB.
SOLAR ARRAY DRIVE	1	5.8"D x 7.7"	14 LB.
SOLAR ARRAY SUN SENSOR	1		
SHUNT REGULATOR	8	7" x 5" x 2.5"	0.8 LB.
BATTERY CHARGER	4	12" x 7" x 4.76"	10.5 LB.
BOOST REGULATOR	4	12" x 7" x 4.76"	10.5 LB.
BATTERY (6-CELL PACK)	12	7" x 5.9" x 4.25"	10.2 LB.
RELAY BOX	1	12" x 7" x 4.76"	5.0 LB.
LOAD RESISTOR INSTL	4	10" x 12" x 1.0"	2.0 LB.

4.4.1 Power Subsystem Design (Continued)

Power subsystem reliability considerations were primarily based on mission life of one year and the general criteria that "no single point failure shall impact capability to complete the basic mission. Based on this criteria, designed redundancy deemed necessary is included and is listed in Table 4.4-7.

TABLE 4.4-7

RELIABILITY CONSIDERATIONS

REDUNDANCY	
SOLAR ARRAY -	INHERENTLY REDUNDANT (SERIES-PARALLEL SOLAR CELLS WITH DIODE ISOLATION FOR EACH SERIES STRING)
SHUNT REGULATORS -	FOUR UNIT PAIRS, SIZED TO PROVIDE REGULATION WITH ONE UNIT PAIR, FAILED FOR MINIMUM SATELLITE LOAD AND MAXIMUM SOLAR ARRAY OUTPUT POWER CONDITIONS
BATTERY CHARGERS -	FOUR UNITS NO MISSION DEGRADATION WITH ONE FAILED UNIT, EACH UNIT CAPABLE OF CHARGING ONE OF TWO BATTERIES
BOOST REGULATORS AND BATTERIES -	FOUR UNITS NO MISSION DEGRADATION WITH ONE FAILED REGULATOR AND/OR ITS ASSOCIATED BATTERY

4.4.2 Solar Array Design. Using the solar array power requirements and Beta angle orientation ranges shown on Figure 4.4-3 and Table 4.4-2, several array concepts were studied. These studies utilized specific solar cell performance characteristics, solar array tilt angles to the sun, and specific array temperatures for given tilt angles.

4.4.2.1 Solar Cell Performance. The performance of an individual solar cell for four different conditions of interest were obtained by shifting the curve in Fig.4.4-4 along the current and voltage axis by calculated amounts. The four conditions are end of life performance and are:

1) Solar array normal to sun and at 66^{o}C
2) Solar array tilted 45^{o} from normal and at 39^{o}C
3) Solar array tilted 30^{o} from normal and at 59^{o}C
4) Solar array tilted 60^{o} from normal and at 20^{o}C

NOTE: The above solar array temperatures vs sun angle are preliminary estimates of the maximum avg. panel temperature during the orbit.

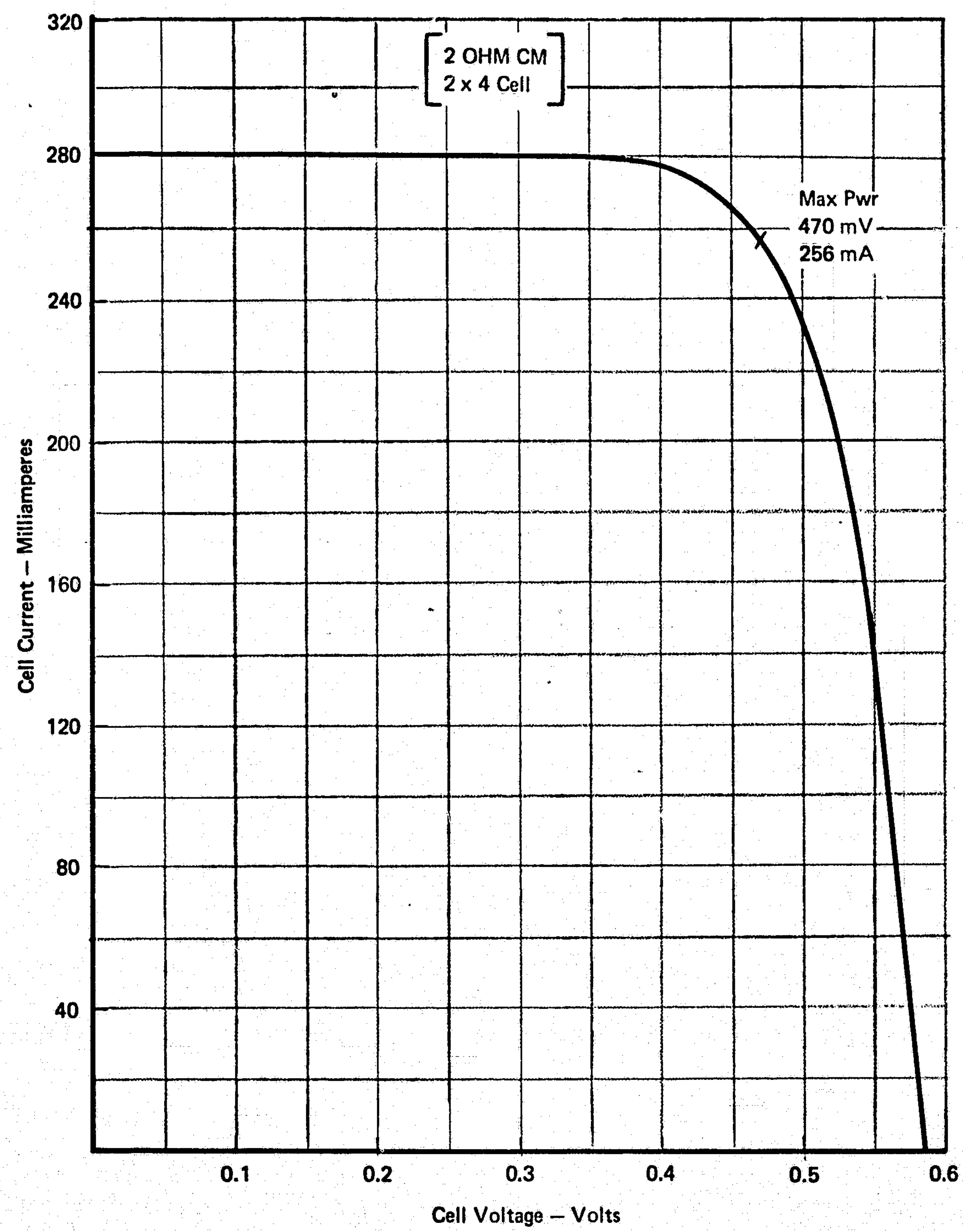

Figure 4.4-4: Solar Cell I-V Characteristics

4.4.2.1 Solar Cell Performance (Continued)

CURVE SHIFTING EQUATIONS

The following equations were used to determine the required voltage and current shifts:

$$\Delta V = (F_V-1)\, V_{oco} + K_V\, V_{oco}\, (T-T_o) + (1-F_I')\, I_{sco}\, R_{sco}$$

$$\Delta I = (F_I-1)\, I_{sco} + K_I\, I_{sco}\, (T-T_o)$$

Where

F_V = Product of design and environmental factors for voltage degradation.

F_I = Product of design and environmental factors for current degradation.

F_I' = Product of design and environmental factors for current degradation except those which are accompanied by voltage degradation (i.e., except module assembly, measurement error, and solar cell charged particle degradation.

K_V = Open circuit voltage temperature coefficient expressed as a fraction of open circuit voltage per degree C.

K_I = Short circuit current temperature coefficient expressed as a fraction of short circuit current per degree C.

V_{oco} = Open circuit voltage of solar cell at standard conditions (28°C, 140 mw/cm^2 AMO).

V_{sco} = Short circuit current of solar cell at standard conditions.

T = Temperature of solar cell

T_o = Temperature of solar cell at standard conditions (28°C)

R_{sco} = Series resistance of solar cell at standard conditions

The following values were assumed for all calculations:

K_V = -0.0028/°C

K_I = +0.00013/°C

T_o = 28°C

V_{oco} = 0.585 volts

I_{sco} = 282 ma

R_{sco} = 0.35 ohms

4.4.2.1 Solar Cell Performance (Continued)

Values for F_V, F_I and F'_I were calculated from the design factors tabulated in Table. 4.4-8. The resulting calculations for these conditions are shown below and the solar cell I-V characteristic corresponding to these conditions are shown in Figure 4.4-4.

Calculations of cell performance for the four cases noted at the beginning of this section follow:

CALCULATIONS

CASE I

$$F_V = (0.980)(0.993)(0.990)(0.980) = 0.945$$

$$F_I = (0.970)(0.994)(0.997)(0.990)(0.976)(0.980)(0.995)(0.966) = 0.904$$

$$F'_I = (0.970)(0.990)(0.976)(0.995)(0.966) = 0.93$$

$$\Delta V = (0.945-1)(.585) + (-0.0028)(.585)(66 - 28^oC) + (1-0.93)(.282)(.35)$$

$$\Delta V = (-.0322) + (-.0622) + (.0069) + -0.087V \text{ and } .585-.087 = .498 \text{ Volts}$$

$$\Delta I = (0.904-1)(.282) + (.00013 \text{ P}/^oC)(66-28)(.282)$$

$$\Delta I = (-.0271) + .00140 = -.026 \text{ and } .282-.026 = .256 \text{ amps}$$

CASE II

$$F_V = .945$$

$$F_I = (0.904)(\text{Cos } 45^o) = (.657)$$

$$F'_I = (0.93)(\text{Cos } 45^o) = (.657)$$

$$\Delta V = (.945-1)(.585) + (-0.0028)(.585)(39^o - 28^oC)+(1-.657)(.282)(.35)$$

$$(-.032) + (-.018) + (.0349) = .015 \text{ and } .585-.015 = .570 \text{ volts}$$

$$\Delta I = (.640)(.282) + (.00013)(.282)(11) = .102 \text{ and } .282-.102 = .180 \text{ amps}$$

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TABLE 4.4-8

SOLAR ARRAY DESIGN FACTORS

	I	V
NON-ENVIRONMENTAL FACTORS		
• CELL EFFICIENCY	1.000	1.000
• COVER INSTALLATION	0.970	1.000
• MODULE ASSEMBLY	0.994	0.980
• MEASUREMENT ERROR	0.997	0.993
ENVIRONMENTAL FACTORS		
• MICROMETEROROIDS	0.990	1.000
• ULTRAVIOLET	0.976	1.000
• TEMPERATURE CYCLING	1.000	0.990
• CHARGED PARTICLES (5.3 X 10^{13})		
SOLAR CELL	0.980	0.980
COVER & ADHESIVE	0.995	1.000
• SEASON FACTOR		
SUMMER SOLSTICE	.966	1.000

SOLAR ARRAY TEMPERATURES

• ARRAY NORMAL TO SUN	66 DEGREES C (AVG)
• ARRAY 30^{o} OFF SUN	59 DEGREES C (AVG)
• ARRAY 45^{o} OFF SUN	39 DEGREES C (AVG)
• ARRAY 60^{o} OFF SUN	20 DEGREES C (AVG)

4.4.2.1 Solar Cell Performance (Continued)

CASE III

F_V = 0.945

F_I = (0.904)(Cos 30°) = .782

F_I' = (0.930)(Cos 30°) - .805

ΔV = (-.032) + (-.0028)(.585)(59-28) + (1-.805)(.282)(.35) =
(-.032) - (.0508) + (.0192) = -.064 and .585 - .064 = .521 volts

ΔI = (0.782-1)(.282) + (.00013)(.282)(31)
(-.0615) + (.00114) = -.061 and .282 -.061 = .228 amps

CASE IV

F_V = 0.945

F_I = (0.904)(Cos 60°) = 0.452

F_I' = (0.930)(Cos 60°) = 0.465

ΔV = (-.032) + (-.0028)(.585)(20-28) + (1-0.465)(.282)(.35)
- (.032) + (.0131) + (.0528) = +.034 and .585 - .034 = .619 volts

ΔI = (0.452-1)(.282)(+ (.00013)(.282)(-8)
(-.1545) - (.000293) = -0.155 and .282 - 0.155 = .127 amps

With the voltage and current values for individual solar cells employed in arrays with sun angles and temperatures for the four cases of interest, as shown in Figure 4.4-5, it is clearly seen that solar array off sun angles of 60° is very near the practical limit for array design when combining array tilt angle of 30° with a Beta angle of 90°, as pictured in Figure 4.4-6.

Calculations for solar cell series/parallel configurations, array areas and total cell quantities are included as follows:

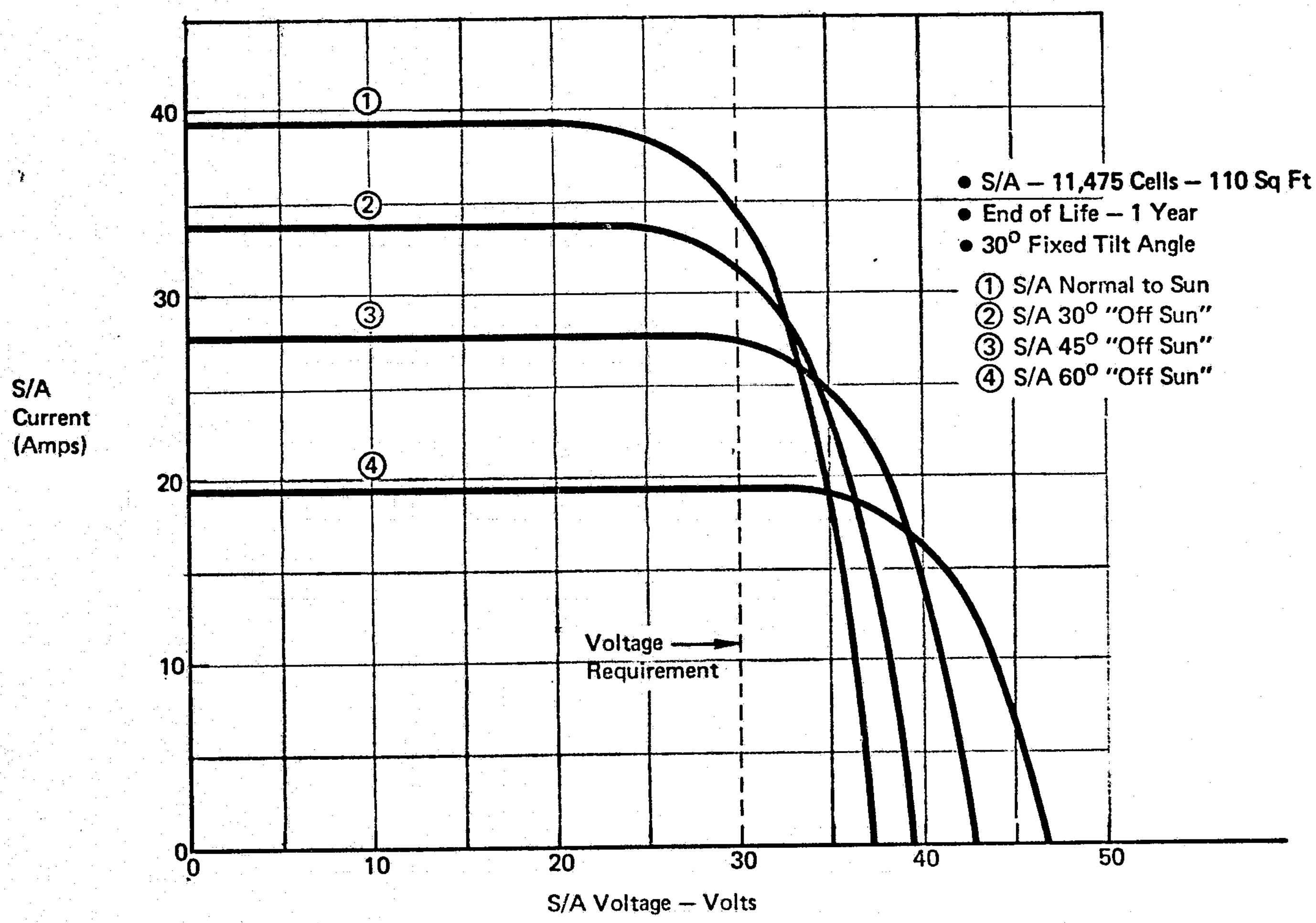

Figure 4.4-5: Solar Array I-V Characteristics

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4.4.2.1 Solar Cell Performance (Continued)

Number of Cells/Area - Tilt Angle = 30°

Series	Output volts required = 30
	29 + 1V Drop for Diode & IR
	30V/.400V = 75 cells
Parallel	905 W/29V = 31.21A
	31.21A/204 ma/cell = 153 cells
Total	153 x 75 - 11,475 cells
Area	11,475 cells @ 105 cells/sq.ft. = 110 sq.ft.

Number of Cells/Area - Tilt = 45°

Series	30V/.440V = 68
Parallel	31.21A/171 ma/cell = 183 cells
Total	68 x 183 = 12,444 cells
Area	12,444 cells @ 105 cells/sq.ft. = 119 sq.ft.

Number of Cells 2-axis art.

Series	30V/360 = 84
Parallel	31.21A/244 ma/cell = 128 cells
Total	128 x 84 = 10,752 cells
Area	10,752 cells @ 105 cells/sq.ft. = 102.4 sq.ft.

Figure 4.4-6 is derived by plotting these data on the power profile previously shown in Figure 4.4-3 and allows optimizing array sizing, shunt regulator sizing and reduced thermal loads by selection of the 30° tilt - fixed, 110 sq. ft. array.

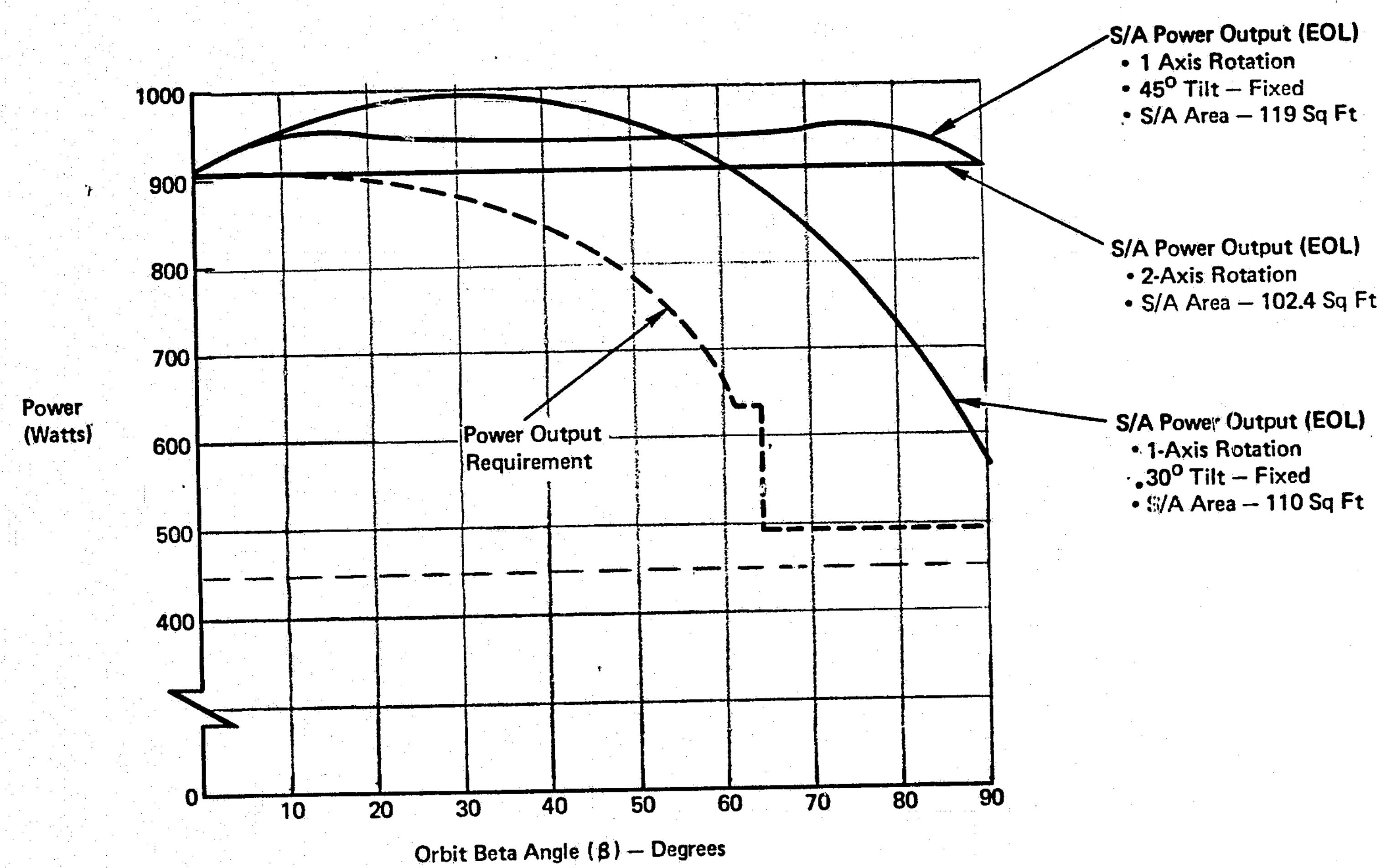


Figure 4.4-6: Solar Array Power vs Orbit Beta Angle

4.4.3 Battery Design. Energy storage required to accommodate connected loads during off sun or occulted periods is provided by four 15 AH nickle-cadmium batteries. Two of these batteries are used on P72-1. Battery capacity for SEASAT is determined primarily by the depth of discharge and cycle life desired for the mission. From Table 4.4-2 the longest occultation period is found to be 35.3 minutes and results in a power drain of 14.7 ampere hours. Recognizing that high levels of discharge have a serious effect on battery cycle life, a maximum of 25% depth of discharge (D.O.D.) is established. From this premise total capacity is determined as follows:

$$14.7 \text{ AH} \div 25\% = 58.7 \text{ ampere hours.}$$

Selecting four 15 AH batteries provides 60 ampere hours of energy and redundancy in event of failure of one battery.

Changes of Beta angle as a function of mission duration is dependent on launch date and is assumed to vary from a potential launch which results in at least some occultations occurring in all orbits for the design lifetime to a potential launch which results in only about 75% of the orbits being occulted. Consequently, the number of battery charger/discharge required is from approximately 3600 to 5400 cycles.

From Figure 4.4-7 we determine that B angles of 45^{o} or larger result in occultation times of 29 minutes or less. Computing for energy levels:

$$29 \text{ minutes} \times 25 \text{ amperes} \div 60 = 12 \text{ AH}$$

$$12 \text{ AH} \div 60 \text{ AH Batt capacity} = 20\% \text{ depth of discharge}$$

In estimating battery life as a function of discharge cycles, results from the Crane tests presented in Figure 4.4-8 are used. These data show that approximately 4000 cycles can be expected with 25% D.O.D. and approximately 5000 cycles with 20% D.O.D. Adverse effects of higher temperature operation are included in the envelope boundaries. Since more than half of the mission orbits will experience depths of discharge less than 20%, the design point of 25% maximum D.O.D. is considered valid.

Additional improvements in cycle life also result from lower operating temperatures, around 60^{o}F. Accordingly modification to the P72-1 charge controller is included to add a trickle charge control function thus preventing excessive overcharge periods.

The P72-1, 15 AH battery is packaged by Hughes in three individual packs of six 15 AH GE cells each. The cells are constrained against bulging from internal pressure by end plates and tension bolts. Each cell has individual thermal fins that are individually bolted to the battery mounting deck for minimum thermal path length and low ΔT. The mounting deck supporting the battery pack installation looks at deep space through a bi-metal controlled louver. This thermal system design has proven very effective in maintaining good thermal balance.

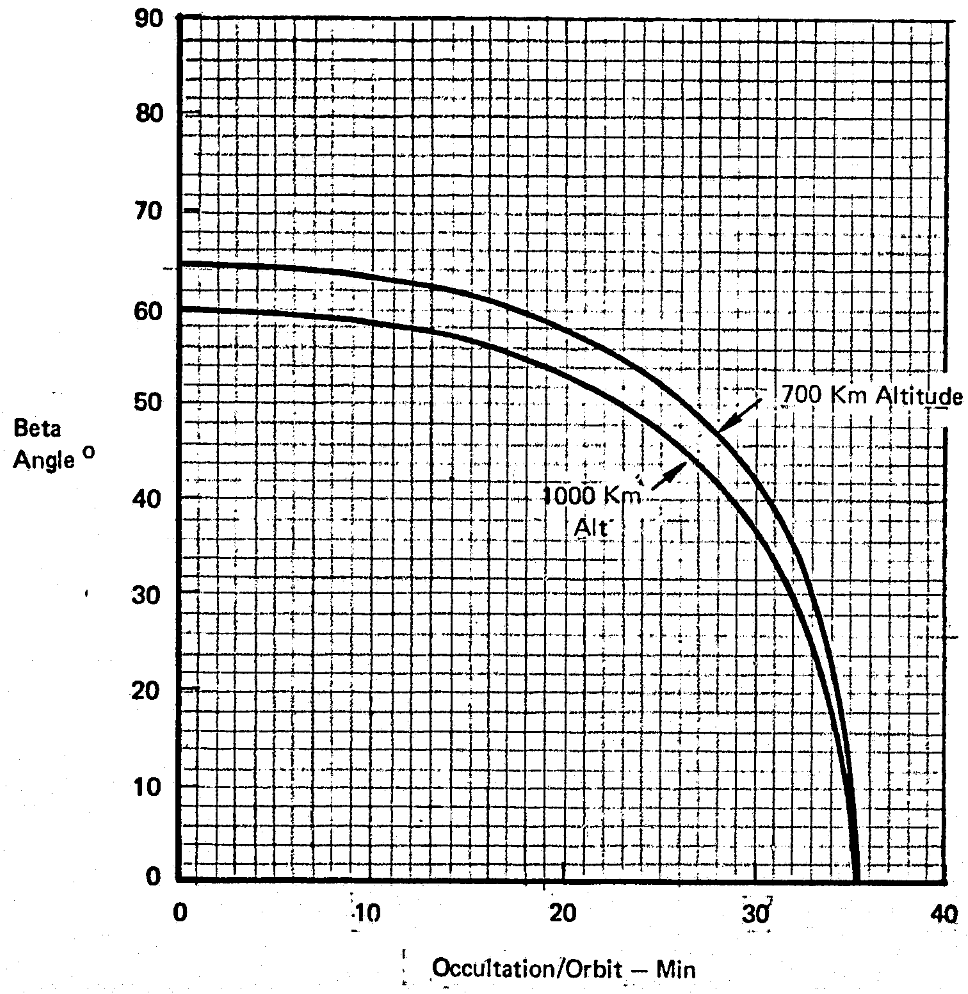


Figure 4.4-7: Occult Time per Orbit vs Orbit Beta Angle

BATTERY CYCLE LIFE VS DEPTH OF DISCHARGE

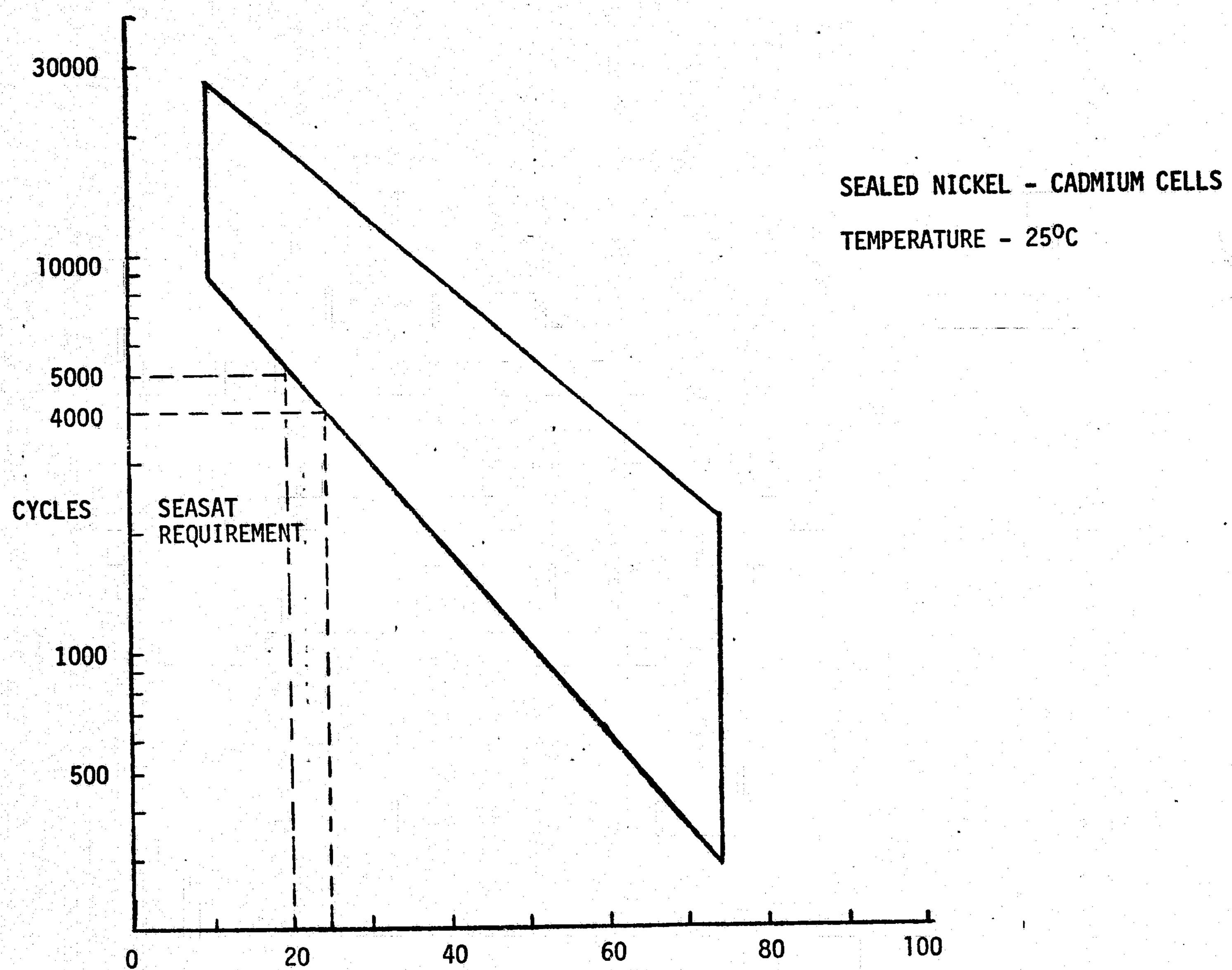


FIGURE 4.4-8

4.4.4 Battery Charger. The battery charger selected for SEASAT is the P72-1 unit with minor modification to the charge current levels and the addition of a commandable trickle charge mode. The charger provides two selectable current limiting modes, 3.5 and 4.7 amperes, changeable by ground command.

With a bus voltage greater than 29 volts and battery voltage 27 volts or less, the charger will charge the battery at the selected current limit. With a bus load greater than the solar array current capability, the charger automatically operates in a voltage limiting mode when the battery voltage or temperature exceeds the limits shown in Table 4.4-9. Interconnections between battery chargers and respective batteries pictured in Figure 4.4-1, show cross strapping so that either one of two chargers may be commanded to charge either of two batteries, providing redundant operation in case of a charger failure.

The trickle charge mode by ground command is provided in the charger for SEASAT to prevent long periods of overcharge and excessive heat generation.

Physical characteristics of the battery charger are listed in Table 4.4-6 and functional characteristics in Table 4.4-9.

TABLE 4.4-9

BATTERY CHARGER FUNCTIONS

INPUT VOLTAGE	27.0 TO 30.0 VDC
CHARGE RATE	3.5 AMPS TO 4.7 AMPS. CONSTANT CURRENT AT 30.0 VOLTS INPUT. PROPORTIONAL CONTROL TO "O" AMPS AT 27 VDC AT BATTERY TERMINALS
OVERTEMPERATURE	CHARGE CURRENT CUT OFF IF BATTERY TEMPERATURE EXCEEDS 95°F AND RESTORES AT BATTERY TEMPERATURE OF 90°F
CHARGE CONTROL	TAPER CHARGE TO PREFERENTIALLY SUPPLY AVAILABLE SOLAR POWER LOADS. AUTOMATICALLY TURN OFF DURING ECLIPSE OR BUS VOLTAGE $< 27 \pm .25$ VDC
COMMANDS	CHARGE EITHER ONE OF TWO BATTERIES SWITCH CHARGE RATE 3.5 OR 4.7 AMPS CHARGER ON/OFF TRICKLE CHARGE ON/OFF

4.4.5 Boost Regulator. The boost regulator selected for SEASAT is used on P72-1 as a battery controller. It operates primarily during occultation or off-sun periods, since the array is sized to carry the spacecraft loads and provide battery charging current under normal conditions. Its function is to provide current to the spacecraft bus from the battery, boosting the voltage in the process to hold the spacecraft bus voltage at 26.25 VDC, although the input voltage to boost regulator may reach a low value of 19 volts. A minor modification is required for SEASAT to provide the additional capability of load sharing among all four operating units, reference Figure 4.4-1. The basic unit as used on P72-1 has this capability for load sharing only between two units.

The boost regulator is enabled automatically when the voltage at the spacecraft bus decreases to 26.25 volts. When enabled it supplies current from the battery to maintain the bus voltage at 26.25 V.D.C. When the solar array output is sufficient to raise the spacecraft bus above the voltage turn-on level, the boost regulator turns off. Use of four boost regulators as diagrammed in Figure 4.4-1 provides redundant capability such that failure of one unit will not adversely affect operation and consequent loss of some mission goals.

Physical characteristics are shown in Table 4.4-6 and functional characteristics are summarized in Table 4.4-10.

TABLE 4.4-10

BOOST REGULATOR FUNCTIONS

REGULATED OUTPUT VOLTAGE	26.25 ± 1.0 VDC
OUTPUT CURRENT	2 TO 13.1 AMPERES
BUS VOLTAGE	24.5 TO 50.0 VDC
BATTERY INPUT VOLTAGE	19.1 TO 26.0 VDC
OUTPUT RIPPLE	< 1.0 VOLT P-P @ MAX LOAD I
LOAD RESPONSE	< 3.0 MS FOR 8.5 AMP PULSED LOAD
TELEMETRY OUTPUTS	CHARGE CURRENT DISCHARGE CURRENT BATTERY VOLTAGE BUS VOLTAGE

4.4.6 Shunt Regulator. The shunt regulator selected for SEASAT during the study period is used on P72-1 and referred to as a voltage limiter. Its function is to bypass or shunt current from the array that is in excess of the amount required for spacecraft loads and battery charging. From the I-V curve characteristics in Figure 4.4-4 the relation of current to voltage can be seen. Thus, as current from the array increases, voltage decreases. Excess current from the solar array is bypassed or shunted through the shunt regulator into resistive load elements that dissipate the resultant heat into space. Voltage on the solar array bus is electronically sensed by the shunt regulators to control the amount of current shunted, and clamp the voltage at the desired level. Referring to Figure 4.4-1 it is noted that two of the four main power feeders between the solar array and the spacecraft bus have switch contacts to provide isolation of portions of the solar array. This isolation feature allows sequential loading to be commanded, thus precluding overload on the shunt regulators at beginning of array life and periods during mission operation where load reduction may be desired. Redundant contacts are used and design detail of connections and operation will be optimized during design phase.

Calculations for shunt regulator system size follow:

Initial assumptions:

- Array is oriented normal to sun
- Winter solstice condition
- Panel temp $<40^{\circ}C$

From solar cell calculations, Section 4.4.2

$$\Delta I = (F_I - 1) I_{sco} + K_I \, I_{sco} \, (T - T_o)$$

$$= (.995-1).282 + (0) = .021$$

$$F_I = (.970)(.994)(.997)(1.0_{env.factors})(1.034)$$

Cell $I_{sc} = .282 + .021 = .303$

Total S/A $I_{sc} = .303 \times 153 = 46.4$ amps

Total Power $\approx 46.4 \times 29V = 1350$ watts

4.4.6 Shunt Regulator (Continued)

From Table 4.4-1 connected spacecraft bus loads are 100 watts ∴ 1350 - 100 = 1250 watts, power required to be shunted if the array were oriented normal to sun, all panels connected to bus and no payload on. Sizing the shunt regulator system for such an adverse transient condition is not efficient design, therefore the following features are included in the system to assure reduced transient power levels.

- o Redundant pairs of isolation contacts in array feeders, ground commanded
- o Array drive commands to provide off-sun normal orientation and tracking

Load resistors for shunted power dissipation are mounted on aluminum heat dissipating panels located on the back sides of the first inboard panel of the solar array on each side of the spacecraft. Location of the shunt regulators and their load resistors on the solar panel structure outboard of the solar array drive has the following advantages:

- o Reduces the number and current rating of slip rings required for power conduction.
- o Significantly reduces the thermal load on the spacecraft bus
- o Simplifies design of the spacecraft electrical distribution and thermal systems.

Physical characteristics of the shunt regulators are listed in Table 4.4-6 and functional characteristics are summarized in Table 4.4-11.

TABLE 4.4-11

SHUNT REGULATOR CHARACTERISTICS

SECTION A VOLTAGE LIMIT	29.25 ± .25 VDC
SECTION B VOLTAGE LIMIT	29.75 ± .25 VDC
RESPONSE TIME	75 MICRO SECONDS WHEN OPERATING IN EITHER SECTION TO 1% OF SET POINT VOLTAGE FOR LOAD CHANGE OF 0.5 AMPS
STANDBY CURRENT	20 MILLI-AMPERES
MAXIMUM DISSIPATION (BOTH SECTIONS)	70 WATTS

Optimization analysis should be conducted to consider reducing the total number of units by increasing size and rating of each unit. Installation simplification and efficiency improvements should be expected.

4.4.7 Solar Array Drive. The solar array drive unit is a single axis, through shaft, direct drive unit with a potentiometer for position feedback and rate control. It utilizes components having a previous space flight history with some repackaging to optimize through shaft and wiring interfaces to the slip rings. In addition to the torque drive unit noted above, a sun sensor with a field of view of $\pm 20^{\circ} \times 90^{\circ}$ is used to provide automatic clockwise or counter-clockwise tracking of the solar array with the sun, in the normal mode of operation. During occultation the solar array drive continues to rotate at the nominal tracking rate during passage through the earth's umbra. The unit is also operatable by command control without use of the sun sensor in either a clockwise or counter-clockwise direction.

Physical characteristics of the solar array drive are shown in Table 4.4-6, functional characteristics are summarized in Table 4.4-12, and function block diagram is shown in Figure 4.4-9.

TABLE 4.4-12

SOLAR ARRAY DRIVE FUNCTIONAL CHARACTERISTICS

TORQUE - BASED ON ARRAY INERTIA OF 330 SLUG FT^2

SLIP RINGS	4 10 AMP POSITIVE POWER CIRCUITS
	4 10 AMP POWER RETURNS
	4 16 AMP SQUIB FIRING CIRCUITS
	4 16 AMP SQUIB RETURN
	2 .02 AMP TEMPERATURE SENSE CIRCUITS
	2 .02 AMP TEMPERATURE SENSE RETURNS
	1 POTENTIOMETER BRUSH
	SPARE CIRCUITS
DRIVE RATES	FIXED POSITION
	FROM 0.0602 DEGREES/SECOND TO 0.0579 DEGREES/SECOND
ACCURACY	$\pm 5^{\circ}$ MAXIMUM WHEN CONTROLLED WITH SUN SENSOR IN CLOSED LOOP MODE
INPUT VOLTAGE	28 ± 4 VDC

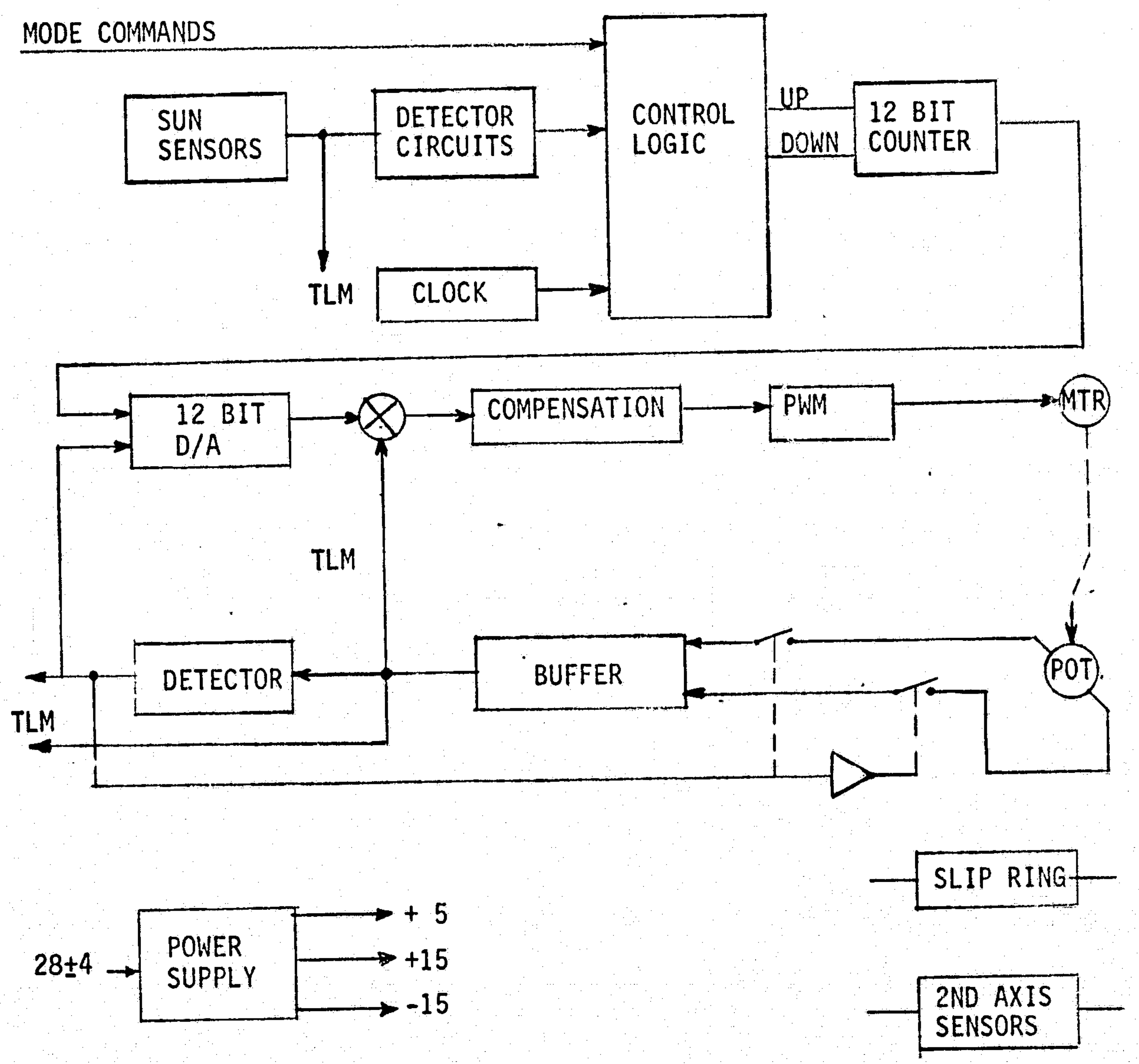


FIGURE 4.4-9

SOLAR ARRAY DRIVE SYSTEM DIAGRAM

4.4.8 Power Relay Box. The power relay box provides the interface points of all primary power switching functions, disconnect and isolation relays, bus load and payload current sensors, and the spacecraft single point ground. The same basic design concept used in the relay box for S3 and P72-1 is used in the relay box for SEASAT.

The power relay box operates in conjunction with the TT&C to accept enable and disable commands and mode change control. Power system functions accessible for ground command through the TT&C are:

- Boost regulator enable/disable
- Solar Array bus tie contactor
- Battery charger commands
  - Charge either one of two batteries
  - Switch charge rate 3.5 or 4.7
  - Charger ON/OFF
  - Trickle charger ON/OFF
- Spacecraft power ON/Ground power AGE ON
- Solar Array Drive orientation
  - Fixed position
  - Drive rate 0.0579^{o}/sec } 11 increments
  - Drive rate 0.0602^{o}/sec } 11 increments
  - High torque mode

4.4.9 Wiring and Connectors. Wiring harness fabrication, connector selection, wire selection and processes are controlled by Boeing document D2-82676-1, Wire Harness Fabrication and Installation - Burner II Flight Vehicle. This document was developed early in the Burner II history to provide design, fabrication and installation criteria for low cost, space flight vehicle wiring. Documented wire bundle grouping requirements and shielding techniques assure a high level of EMI noise immunity and have demonstrated faultless operation in many space vehicles built by Boeing over the past ten years.

5.0 PAYLOAD MODULE

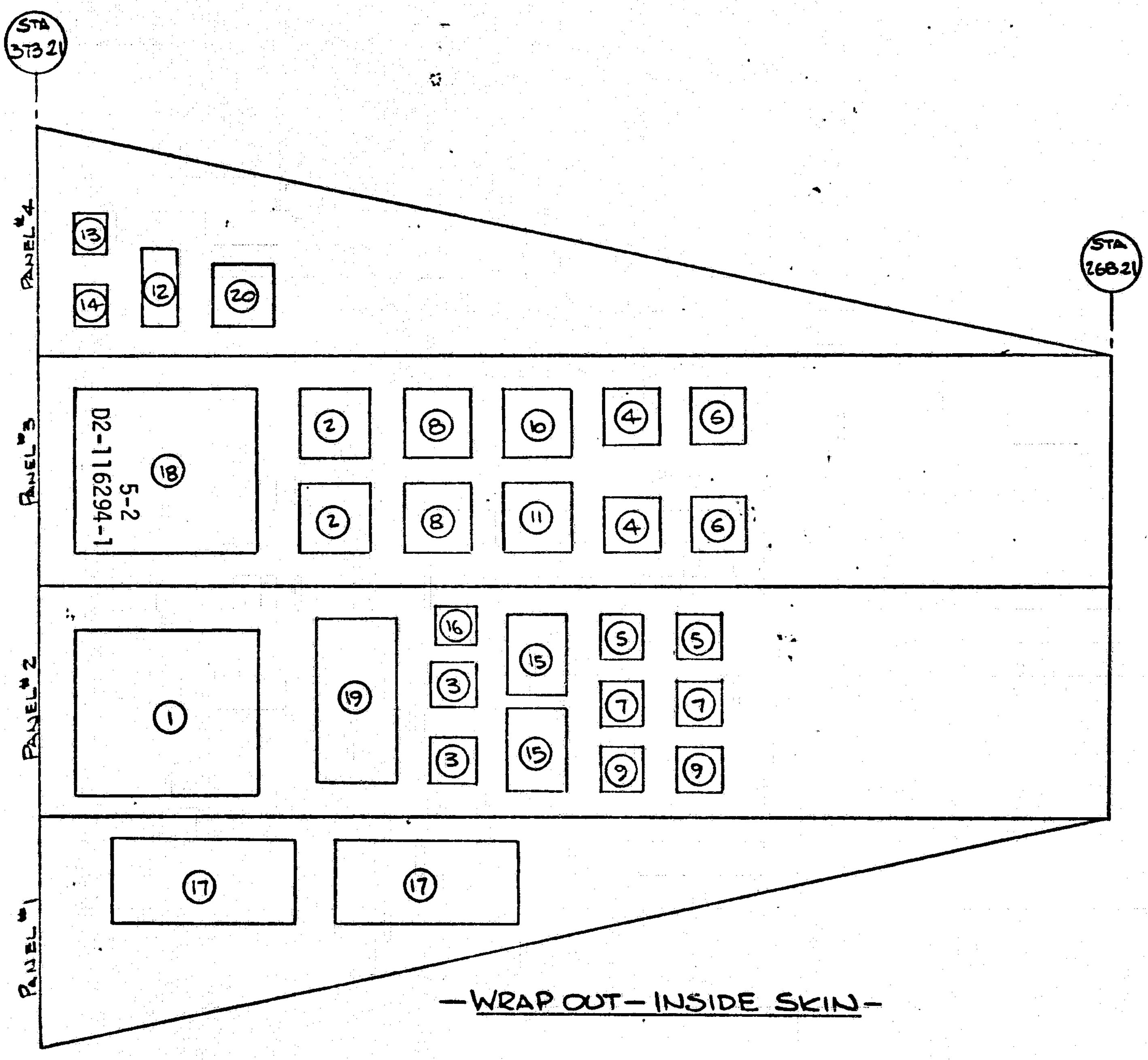
5.1 CONFIGURATION

The proposed "Payload Module" structure is approximately (105) inches long. The aft end of the structure which interfaces with the "Bus Module" is hexagonal in shape and this transitions to a triangular section at the forward end. Conventional aluminum structure is used to effectively meet all requirements for stiffness, strength and equipment support. The module is completely "skinned" and utilizes extruded longerons for the primary load paths. Secondary structural supports are provided for the equipment located within the module.

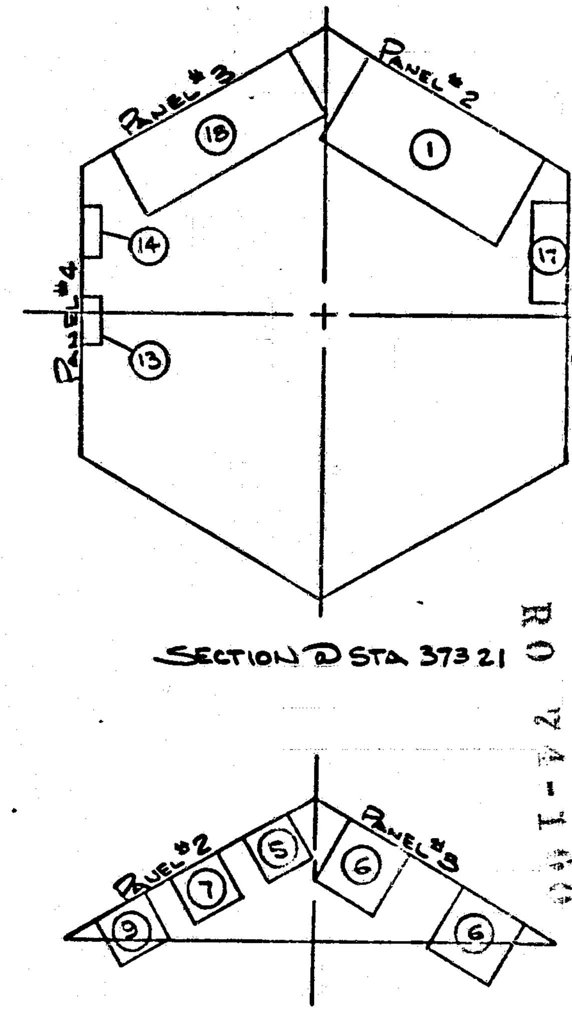
Eight (8) antennas are mounted on the "Payload Module" - (5) stick antennas, (1) one meter fixed dish antenna, (1) one meter articulated dish antenna, and (1) conic doppler beacon antenna. All other payload equipment is mounted within the module on the vehicle skin as shown in Figure 5.1-1. The (5) stick antennas are stowed in a forward and aft position during launch and boost and are hinged at the forward end to accommodate deployment - see the "On Orbit" configuration, Figure 3.1-2. Deployment of the stick antennas will be accomplished with an actuator identical to that which is used to deploy the solar panels and some of the antennas on both the Lunar Orbiter and MVM '73 space vehicles. Design of the equipment installation considered such items as accessibility for test and replacement, spacecraft center of gravity location and thermal control. Thermal control of the "Payload Module" is accomplished by the use of louvers, paint and blankets. Alignment of sensors and antennae having low accuracy ($.5^{o}$) requirements will be accomplished by aligning the mounting surfaces with respect to the spacecraft reference axes.

For sensors and equipment that require precision installations, optical alignments will be made with theodolites. Optical mirrors mounted to reference surfaces on the face of the equipment provide a standard method of accomplishing this task. Shims and locators are used to obtain alignment and insure the alignment is maintained.

The payload Module design is similar to that developed for Burner II and STP P72-1 hardware and can be designed and developed with a high level of confidence. All structural concepts and subsystem features are flight proven.



-WRAP OUT-INSIDE SKIN-



SECTION @ STA 373.21

SECTION @ STA 268.21

RO 74-160

-PAYLOAD MODULE, SEASAT BASELINE CONFIGURATION-

SCALE 1:10

FIGURE 5.1-1 SH 1 OF 2

REF NO	NOMENCLATURE	QTY
1	ALTIMETER	1
2	6.6 GHz RFA	2
3	6.6 GHz IF/VIDEO	2
4	18 GHz RFA	2
5	18 GHz IF/VIDEO	2
6	22 GHz RFA	2
7	22 GHz IF/VIDEO	2
8	37 GHz RFA	2
9	37 GHz IF/VIDEO	2
10	TLM/COMM & TIMING	1
11	POWER SUP.	1
12	FREQUENCY SYNTH	1
13	162 MHz P.A.	1
14	325 MHz P.A.	1
15	R.F. SWITCH BOX	2
16	ALTIMITER PREPROCESSOR	1
17	LOUVER	2
18	SCATTEROMETER	1
19	V & IR	1
20	DOPPLER BEACON OSCILLATOR	1

– PAYLOAD MODULE, SEASAT BASELINE CONFIGURATION –

FIGURE 5.1-1 SH 2 OF 2

D. Witton
4/19/74

5.2 SENSOR COMPLEMENT

The baseline configuration sized and priced by Boeing under this study includes the following Payload Sensor Subsystems:

Compressed Pulse Radar Altimeter
Microwave Scatterometer
Four Frequency Microwave Radiometer
TRANET Doppler Beacon
Visual and Infrared Radiometer

The change of direction indicating the need to include the Imaging Radar in the SEASAT-A was not received sufficiently early to include it in the configuration or the pricing of the baseline. Section 10 discusses the impact to the Payload Module, and S/C Bus design of incorporating the Imaging Radar.

In establishing the telemetry subsystem, only the three sensor operating modes defined by JPL letter 627-LA:kl, dated 9 January 1974, were considered. It is recognized that, prior to the hardware phase, sensor operating modes will have to be expanded to provide for operation of only one or two sensors and even a mode wherein no Payload Sensors are turned on.

Each of the above sensor subsystems included in the baseline are discussed below, reflecting our understanding and assumptions.

5.2.1 Compressed Pulse Radar Altimeter. The characteristics of the altimeter which impact the configuration of the Payload Module or S/C Bus are listed and/or discussed below. These characteristics or parameters were obtained from various documentation provided or were assumed in order to configure the S/C.

Frequency: 13.9 GHz ± 180 KHz

Weight: 45 Kg

D.C. Power: 125 Watts average - No figures were found in the documentation indicating the peak d.c. power requirements. A figure of 10 KW is assumed.

Date Bit Rate:

Mode A: 13 Kbps

Mode B: 300 bps - The purpose of the 300 bps mode is not thoroughly understood. The TT&C Subsystem (Section 4.3) as presently envisioned would not change telemetry modes between Modes A and B, and could handle the 13 Kbps rate under all conditions. If the 300 bps data is required on the ground, it could be transmitted in addition to the 13 Kbps data.

5.2 SENSOR COMPLEMENT (CONTINUED)

It has been assumed in configuring the telemetry subsystem that the altimeter will store its 13 Kbps and 300 bps data in registers (e.g., 64 bit capacity) to be read out by the telemetry at a 36 Kbps rate.

Preprocessing: The Maximum Likelihood processing capability is to be provided by the altimeter since the S/C Bus, as presently configured, does not have need for a general-purpose computer. Before or during the hardware phase the need for a S/C Bus Computer will be re-evaluated when payload telemetry and commands are better defined.

5.2.2 Microwave Scatterometer. It is recognized that the Scatterometer configuration is probably more fluid at this point than any of the other sensors; however, the characteristics or parameters were obtained from various documentation provided or were assumed in order to configure the S/C.

Frequency:	9 - 14 GHz
	A single frequency TBD; however, 14 GHz was assumed in the baseline.
Weight:	75 Kg
Size:	50 x 10^3 cm^3
	Two transmitters and receivers (horizontal and vertical) are understood to be included.
D.C. Power:	150 watt average.
	No good figures were found in the documentation for the peak d.c. power requirements. Based upon a 15% duty cycle, 1000 watt peak was assumed. See discussion in Para. 5.2.2.1.
Data Bit Rate:	500 bps
	It has been assumed in configuring the telemetry subsystem that the scatterometer will store its 500 bps in registers (e.g., 64 bit capacity) to be read out by the telemetry at a 36 Kbps rate.
Antennas:	5-Stick Arrays (See 5.3.4)

5.2.2.1 Discussion. There appears to be a serious discrepancy in the scatterometer power requirement established for the baseline. JPL configuration A3 indicates 150-200 watts (d.c.) which is compatible with the 150 watts of JPL letter 627-LDA:vr, dated 1 March 1974. With reference to Figure 22 of the Preliminary Scatterometer Specification, dated 6 November 1973, the r.f. power requirement for a $0.5^{o} \times 55^{o}$ stick array can be extrapolated to be 140 watts average. Assuming a 30% efficiency, the d.c. input required would be 467 watts average for the transmitter only.

Not knowing the validity of the preliminary specification, the S/C baseline described in this report is in accordance with the HPL established requirement of 150 W d.c.

5.2.3 Four Frequency Microwave Radiometer. The passive radiometer characteristics which impact the configuration of the Payload Module or the S/C Module are listed and/or discussed below. These characteristics or parameters were obtained from various documentation provided or were assumed in order to configure the S/C.

Frequencies: 6.6 GHz
18. GHz
22. GHz
37. GHz

Size and Weight:

R.F. Assy.	5100 cm^3	5.2 Kg
R.F. Assy.	5100 cm^3	4.1 Kg
2 R.F. Assys.	2600 cm^3 (ea)	3.0 Kg (ea)
4 IF/Video Assys.	1300 cm^3 (ea)	1.3 Kg (ea)
T/M & Com. Assy.	5100 cm^3	4.0 Kg
Power Supplies	5100 cm^3	5.8 Kg

Two receivers (horizontal and vertical) are understood to be included for each frequency.

D.C. Power: 45 Watts

Data Bit Rate: 2 Kbpx

It has been assumed in configuring the telemetry subsystem that the radiometer will store its 2 Kbps data in registers (e.g., 64 bit capacity) to be read out by the telemetry at a 36 Kbps rate. Engineering (housekeeping) data will be sampled on an individual measurement basis.

Antenna: 1-Meter diameter parabola - scanning (See 5.3.2)

5.2.4 TRANET Doppler Beacon. The beacon operationally used by the Navy Navigation Satellite System is to be installed in the Payload Module. Its characteristics which impact the configuration of the Payload Module and the S/C Bus are listed and/or discussed below. These characteristics were obtained from documentation provided by JPL letter 627-LDA:vr/2.13.4, dated 13 February 1974, or were assumed in order to configure the S/C.

Frequencies: 162 MHz
342 MHz

Size and Weight:

Oscillator:	8.8 dia. x 16.5 cm	1.45 Kg
Freq. Synth.	8.7 x 18.1 x 3.5 cm	.59 Kg
162 MHz P.A.	8.7 x 9.8 x 3.5 cm	.32 Kg
342 MHz P.A.	8.7 x 9.8 x 3.5 cm	.32 Kg

D.C. Power: 5.5 Watts

Data Bit Rate: Engineering measurements only.

Antenna: Dual frequency quadrifilar helix (See 5.3.5)

5.2.5 Visual and Infrared Radiometer. The V&IR radiometer currently selected for SEASAT-A is an AVHHR system modified to an SSTIR. The SSTIR characteristics understood to apply to the modified AVHHR unit that impact the Payload Module and the S/C Bus are listed and/or discussed below. These characteristics were obtained from the SSTIR Fact Sheet dated 1 November 1973 and verbal inputs from JPL or were assumed in order to configure the S/C

Viewing Angle: Nadir ± 54^o

Attitude Control: 0.5^o

Attitude Determination: 0.1^o - 0.2^o

Weight: 18 Kg

D.C. Power: 20W

Data Bit Rate: 18 Kbps

It has been assumed in configuring the telemetry subsystem that the V&IR will store its 18 Kbps data in registers (e.g., 100 bit capacity in a multiple of the digitizer word size) to be read out by the telemetry at a 36 Kbps rate.

Processing: Digitizing (A/D conversion) and data compression (line sampling to reduce bit rate to 18 Kbps) is understood to be accomplished by the V&IR sensor.

5.2.5 Visual and Infrared Radiometer (Continued)

Although the SEASAT-A baseline as presented herein is based upon a V&IR bit rate of only 18 Kbps, higher data rates could be considered prior to the hardware phase when more is understood about UWG requirements, operating modes, etc. For example: Little S/C Bus impact (including cost) would be involved in transmitting the 465 Kbps data in real-time on the high-rate subcarrier while in sight of a ground station after stored data playback.

5.3 SENSOR ANTENNAS

The sensor antennas required for the defined mission of the SEASAT-A satellite constitute a major Payload Module Subsystem. It is this antenna subsystem which determines the overall Spacecraft as well as the Payload Module configuration. Prior to the receipt of JPL letter 627-LDA:vr, dated 1 March 1974, Boeing had evaluated the various antenna configurations suggested by the JPL configurations A1 through A4, B1 and B2. (Table 5.3-1 outlines these configurations for reference. In selecting the particular antenna for each sensor, every effort was made to provide the data considered to be preferred by the User Working Group (UWG). This understanding of UWG preference was obtained, primarily, through the interpretation of available documentation provided on each sensor.

Hardware considered a part of the Antenna Subsystem includes: (1) the antennas (e.g., reflectors and r.f. feeds), (2) waveguide and coax, (3) r.f. movable joints, and (4) r.f. switching.

In selecting sensor antenna sizes, no attempt was made to conduct r.f. link analyses to establish gain requirements. Antenna sizes shown in the Preliminary SEASAT-A Program Plan, dated 26 October 1973, were accepted as being satisfactory without gain optimization to approach theoretical figures. Antenna costs were based on achievement of nominal gain from the antennas.

The rationale for the antennas selected for the Boeing C1 and C2 configurations will be discussed under "Trades" of each sensor's antenna where C1 and C2 differ from the baseline and where considered a significant option.

5.3.1 Compressed Pulse Radar Altimeter. The basic requirements for the altimeter antenna are:

Frequency:	13.9 GHz
R.F. Power:	2.5 KW peak
Pulse Length:	3 ns
Look Angle:	Nadir (0^o Cone)
Scan:	Fixed

5.3.1.1 Baseline Antenna. The one (1) meter diameter parabolic reflector suggested in JPL configurations A1, A4, B1 and B2 was selected for the baseline configuration. For this study, Boeing would propose an aluminum honeycomb core with either graphite-epoxy or aluminum skins since flight proven designs from VO'75, MM '71, and MVM '73 are available. The MM '71 reflector has a 40" diameter whereas the MVM '73 and VO '75 reflectors would have to be modified.

Subsequent to this evaluation and selection effort, JPL elected to provide the antenna as GFE along with the altimeter. Therefore, no cost estimate or discussion of a selected antenna is included in this report.

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TABLE 5.3-1

JPL SUGGESTED ANTENNA CONFIGURATIONS

	A1	A2	A3	A4	B1	B2
ALTIMETER	1M PARABOLIC	1M PHASED ARRAY	1M PHASED ARRAY	1M PARABOLIC	1M PARABOLIC (1)	1M PARABOLIC (4)
SCATTEROMETER	2M PARABOLIC (CONICAL SWEEP) 1.5M PARABOLIC (CONICAL SWEEP) 1.5M PARABOLIC (CONICAL SWEEP) 1.5M PARABOLIC (FWD. SWEEP)	2M PHASED ARRAY (CONICAL SWEEP) 1.5M PHASED ARRAY (CONICAL SWEEP) 1.5M PHASED ARRAY (FWD. SWEEP)	5 STICK ARRAYS ($0.5^0 X 50^0$) CLOCK ANGLES: 45, 135^0, 225^0, 315^0, AND 0^0	2M PARABOLIC (40^0 CONICAL, 1^0 CONICAL, & FWD SWEEPS) (3)	12 STICK ARRAYS (0.5^0 x 13^0) CLOCK ANGLES: 45^0, 135^0, 225^0, AND 315^0 CONF ANGLES: 6.5^0, 30^0, & 45^0.	2M PARABOLIC (2) (FWD. SWEEP)
RADIOMETER	2M PARABOLIC (CONICAL SWEEP) 1M PARABOLIC (CONICAL SWEEP)	2M PHASED ARRAY (CONICAL SWEEP) 3- .8M PHASED ARRAYS (CONICAL SWEEP)	2M PARABOLIC (CONICAL SWEEP)	2M PARABOLIC (3)	1M PARABOLIC (1)	1M PARABOLIC (4) 2M PARABOLIC (2)

(1) JOINT USE - ALTIMETER AND RADIOMETER

(2) JOINT/TIME SHARED USE - SCATTEROMETER AND RADIOMETER (PARTIAL)

(3) TIME SHARED - SCATTEROMETER AND RADIOMETER

(4) JOINT USE - ALTIMETER AND RADIOMETER (PARTIAL)

5.3.1.2 Trades. In selecting the parabolic reflector for the altimeter, a phased array was considered. However, prior to completing the evaluation and a preliminary design, the C.T. Swift report dated 30 November 1973 was found in the document "Antenna Study for SEASAT" dated 2 December 1973. Based upon this report, which indicated that a 1 meter phased array for the altimeter would not work, further trade study effort was stopped.

During later SEASAT-A phases, additional study is considered necessary to establish minimum gain requirements; to evaluate other available designs and to optimize the type of feed if the antenna is not to be GFE.

5.3.2 Four Frequency Scanning Microwave Radiometer. The basic requirements for the passive radiometer antenna(s) are:

Frequencies:	6.6, 18, 22, and 37 GHz
Look Angle:	45^o Cone
Scan:	$\pm$ 35^o across track

5.3.2.1 Baseline Antenna System. With the receipt of JPL letter 627-LDA:vr, dated 1 March 1974, the Microwave Radiometer antenna baseline was established as the Nimbus G fixed multifrequency feed with a 1 meter diameter offset parabola. Due to the NASA procurement schedule for Nimbus G, no data can be made available to industry for evaluation or pricing of this antenna system. Therefore, for the purposes of this SEASAT-A study, it is understood that JPL will estimate the subcontractor costs for the Nimbus G type antenna system.

5.3.2.2 Trades. Since the details of the Nimbus G antenna cannot be disclosed at this time and the rationale for its selection as the baseline is unknown, a discussion will be provided here of the Boeing selected baseline. Configuration trades discussed in subsequent paragraphs will generally be presented with reference to this Boeing selected baseline.

Trade 1 - Boeing Selected Baseline. The requirement for an Instantaneous Field of View (IFOV) of less than 100 Km x 100 Km at all frequencies was accepted as firm and any violation of this maximum foot-print was considered to degrade the data beyond the UWG's acceptance. With this IFOV requirement, two parabolic reflector antennas were selected as the study baseline configuration. This is in agreement with an option indicated in JPL configurations A1, A3, and A4. A two (2) meter reflector with a C-Band, E and H plane fed circular feed horn satisfies the 6.6 GHz requirement with a 76 Km x 44 Km IFOV at 1000 Km altitude. A separate 80 cm reflector with a multi-frequency feed, such as an Octave-Bandwidth feed horn, fed by two waveguide systems (horizontal and vertical polarizations) will handle the 18, 22, and 37 GHz signals.

Two receivers per frequency (horizontal and vertical polarization) and their associated electronics would be mounted on the articulating structure associated with their respective antennas to eliminate the problems of carrying r.f., video, or analog sensor data across the rotating joint. This approach will improve the probability of getting good surface data but will probably increase the complexity and duplication of some of the sensor electronics (e.g., the telemetry and control unit). The articulation mechanisms would be of existing designs modified only to accommodate the $\pm 35^{o}$ range and the Radiometer slip ring requirements for power, command, and telemetry data signals.

Both antennas have a 45^{o} cone look angle and scan $\pm 35^{o}$ sinusoidally across-track either forward or aft depending on S/C flight attitude. A reversal of flight attitude will occur once every six months, based on sun position, and was necessary to reduce solar cell area.

Trade 2 - Single Four Frequency Reflector. Under-illumination of the 2 meter reflector by the three higher frequencies was abandoned due, primarily, to pattern blockage by the larger 6.6 GHz feed and the multiple coax/waveguide runs to the feeds. Reflector shaping techniques might decrease the blockage but undoubtedly require a non-space proven antenna and be much more expensive. The use of a single reflector of 80 cm or 1 meter diameter was not considered due to the obvious violation of the 100 Km x 100 Km IFOV at 6.6 GHz from 1000 Km and the JPL apparent opposition to a 1 meter x 2 meter antenna considered in Boeing configurations C1 and C2 (see Trade 5 below).

Trade 3 - Phased Arrays. The mounting of four (4) radiometer phased arrays, as suggested by JPL configuration A2, on the payload module along with the other SEASAT-A antennas creates a very difficult configuration problem. This approach could require deploying a sizable articulating structure containing a part of the antenna farm.

Due to the work previously accomplished by APL (C. C. Kilgus) and Aerojet General (J. Fujioka) during Phase A, no attempt was made to size or price phased arrays for the radiometer. The size and cost figures in the C. C. Kilgus memo of August 17, 1973 for the phased arrays are probably sufficiently accurate due to the Aerojet experience; however, the 55 Kg and $1.4M figures for the 2 meter dish appear extremely high. It is considered that even with two reflector antennas (2m and 0.8m), the phased array approach would be at a disadvantage from both the weight and cost standpoint.

Trade 4 - Unfurlable 2 Meter Reflector. The unfurlable reflector built by Radiation, Inc. will provide the performance required at 6.6 GHz. Its use in lieu of the 2 meter rigid reflector would permit the use of the smaller Burner II shroud without increasing the 6.6 GHz IFOV by an elliptical reflector (1m x 2m). With the selection of the Atlas/Burner II launch vehicle, the Radiation 2m dia. unfurlable or an elliptical reflector would be recommended for consideration if the smallest possible IFOV is the objective.

Trade 5 - Elliptical 6.6 GHz Antenna. An elliptical (1 m x 2 m) reflector was considered in the Boeing C1 and C2 configurations for the 6.6 GHz based upon an IFOV of "less than 100 Km x 100 Km". The IFOV figures in Table 2 (1000 Km altitude) of the Four Frequency Microwave Radiometer Spec. dated 6 November 1973 show 76.2 Km in-track and 43.8 Km cross-track for a 2 meter circular antenna (1.62^{o} beam width). Doubling the beam width by halving the reflector size in the cross-track direction would still meet the 100 Km x 100 Km IFOV.

The advantage of a 1 m x 2 m reflector is evident in simplifying the deployment (in the C1 and C2 configurations) when a Delta shroud is used and permitting a non-unfurlable 6.6 GHz antenna that meets the IFOV in the smaller Burner II shroud. It should be noted that deployment of the 2 meter dia. antenna is not required in the Boeing baseline configuration (Trade 1) due to the use of the stick arrays in lieu of two 1.5 meter parabolic reflectors for the Scatterometer.

5.3.3 Microwave Wind Scatterometer. In accordance with the direction of JPL letter 627-LDA:vr, dated 1 March 1974, the scatterometer antenna baseline was established as essentially that shown in the JPL A3 configuration. From this baseline, the sensor antenna requirements are:

Frequency:	9-14 GHz (Single Freq. TBD)
Look Angle:	0^{o} to 55^{o} Cone;
	45^{o}, 135^{o}, 225^{o}, and 315^{o} Clock;
	and 0^{o} Clock for calibration.

For this study, the assumption has been made (based upon information received from JPL) that the sensor will have two transmitters and two receivers (one set for horizontal and one for vertical polarization). Transmit/Receive would be H/H and V/V with no requirement for H/V or V/H.

5.3.3.1 Baseline Antenna System. The stick array antenna configuration of the JPL A3 approach is considered to be approaching the optimum desired by the UWG. Within the general stick array approach, there are several options which should be considered prior to the hardware phase. These will be discussed under 5.3.3.2, Trades.

Prior to the receipt of the JPL directed baseline, Boeing was proposing five 0.5^{o} x 20^{o} stepping stick arrays. The Payload Module Configuration now includes the five 0.5^{o} x 55^{o} fixed (after deployment) stick arrays. Figure 5.3-1 shows the deployed arrangement of the five sticks.

To maintain the polarization purity required by the sensor, it seems appropriate to provide separate antennas (horizontal and vertical polarization) on each stick array. This configuration also provides the advantage of being able to transmit one polarization and receive both polarizations separately (if this should become a desired mode of operation).

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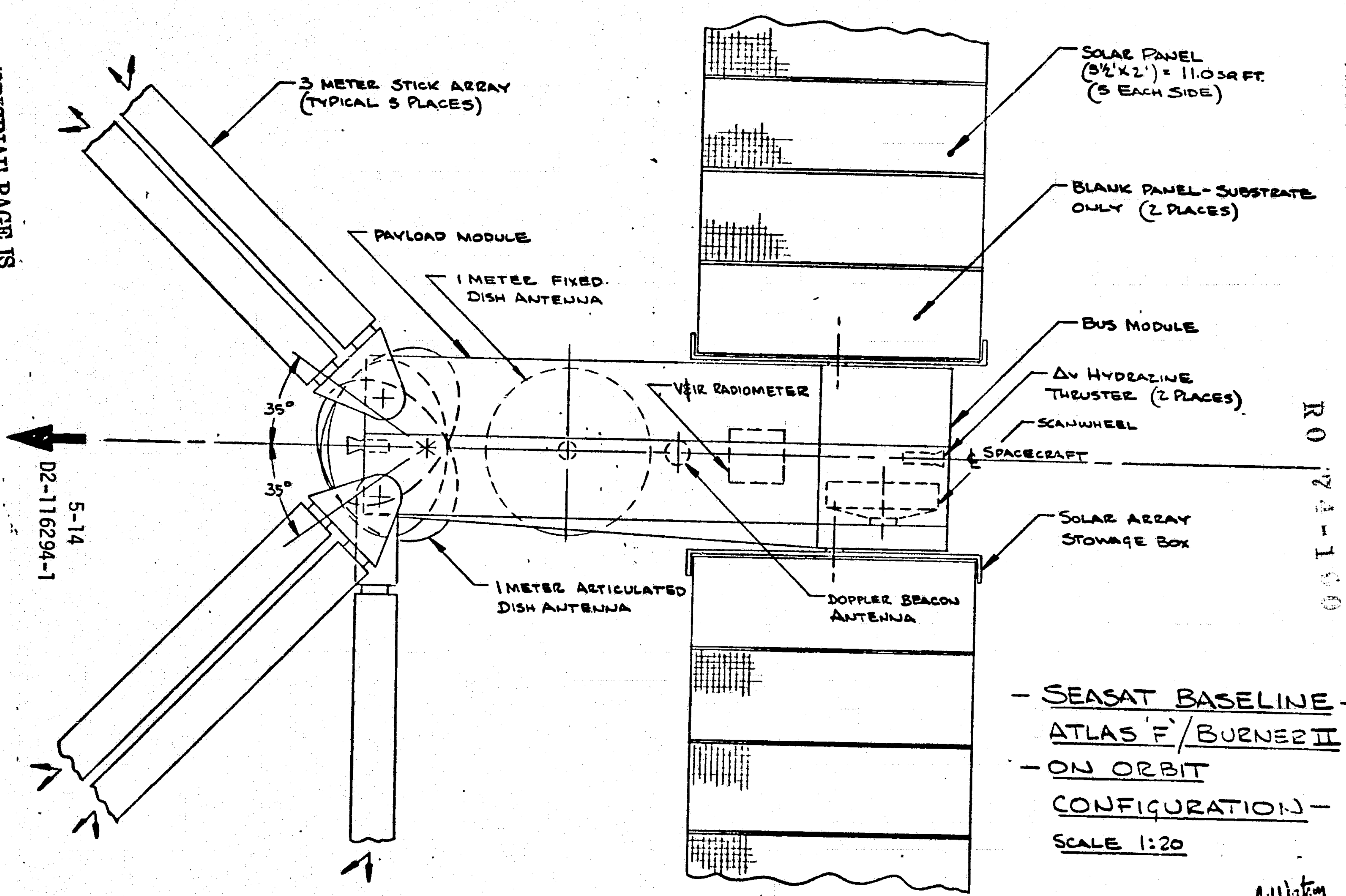


FIGURE 5.3-1

With the $0.5^o \times 55^o$ beam, Figure 5.3-2 shows the IFOV patterns and a typical sequence of viewing. The involvement of the calibration antenna in the normal viewing sequence is as requested by JPL; however, this requirement should be re-evaluated prior to the hardware phase due to the anticipated complexity and insertion loss of the r.f. switching. A switching scheme is shown in Figure 5.3-3 and discussed in paragraph 5.3.3.3.

5.3.3.2 Trades. The Boeing selection of the stick array configuration centered about an attempt to provide what appeared to approach the optimum system from the sensor data standpoint; while, simultaneously, satisfying satellite configuration constraints.

Trade 1 - Fixed-Vs-Stepped Sticks. The selection of stick arrays appeared to be the most logical configuration to pursue after the other types of antennas had been evaluated (See Trades 2 through 4).

Of the stick array approaches possible (fixed and various sizes of stepped arrays), Boeing selected five $0.5^o \times 20^o$ stepped sticks. It was recognized that the stepped sticks had the disadvantage of requiring stepping mechanisms with a high duty cycle. However, even if all the stepping mechanisms should fail, the sensor would still have a view of a swath width of at least 500 Km (1000 Km altitude) across-track both fore and aft. The outstanding advantage of the stepped sticks is the reduced d.c. and r.f. power required (e.g., the 55^o sticks will require 4.4 dB more r.f. power than the 20^o sticks).

The prime reason for not selecting the fixed $0.5^o \times 55^o$ sticks was our concern for the power requirements (both r.f. and d.c.) resulting from the broader beam antenna. At this stage of sensor development (based upon the visibility provided) the sensor d.c. power requirements seemed extremely soft. This could seriously impact the cost of the power subsystem. See paragraph 5.2.2.1.

Trade 2 - Parabolic Antennas. Two configurations were suggested (A1 and A4) utilizing parabolic antennas.

CONFIGURATION A1 requires two 1.5 meter reflectors scanning across track (fore and aft); one 1.5 meter reflector scanning along track for calibration; and a 2 meter reflector scanning 360^o in a 1^o cone. The third 1.5 meter antenna could be deleted by periodically using the forward across track antenna for calibration with resultant complexity of the drive mechanism.

Either the three or four reflector arrangement creates severe Payload Module Configuration problems such as stowage within the shroud, deployment and unobstructed viewing. Even if these configuraton problems could be overcome, the sensor operational sequence to provide the required surface coverage would become quite complicated when compared with the stick arrays.

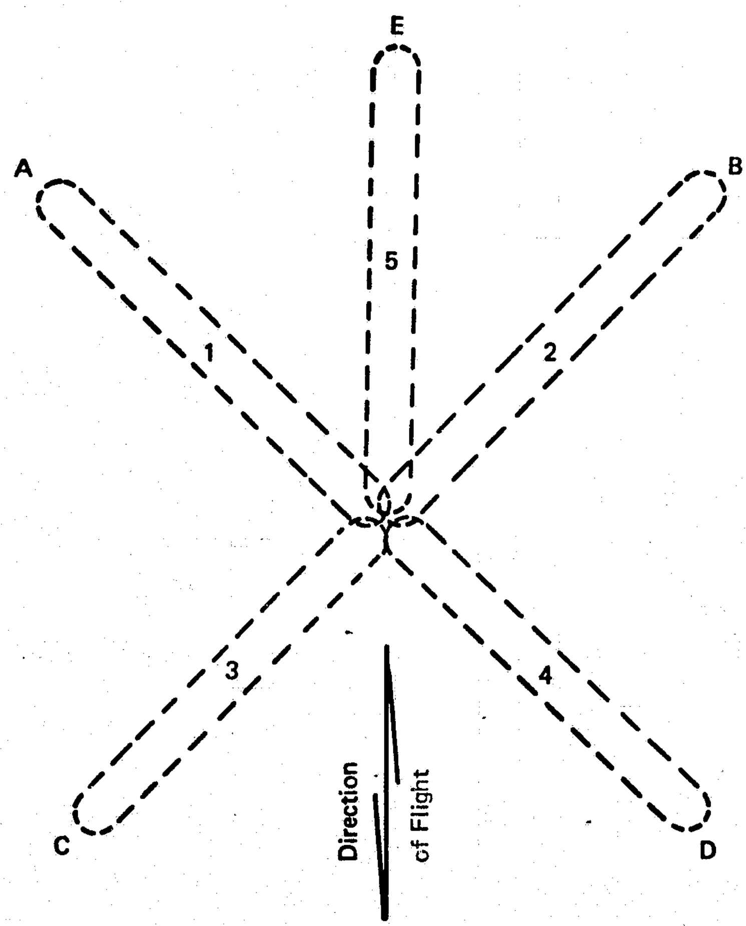
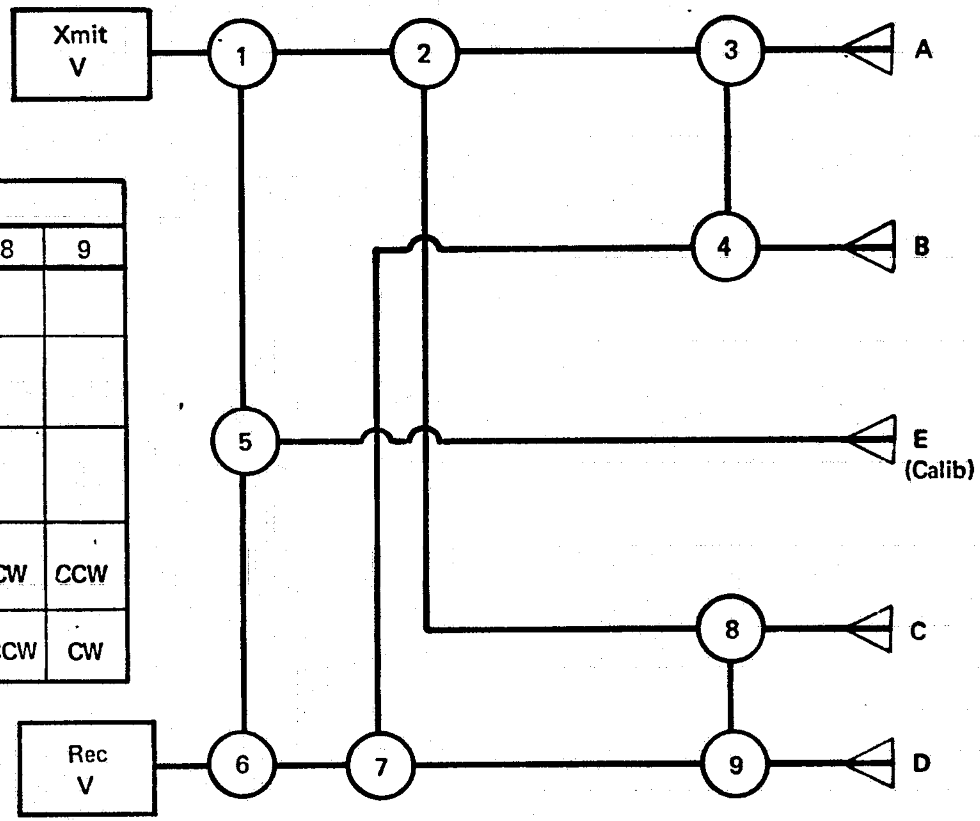


Figure 5.3-2: Scatterometer Antenna Pattern



Ant.	SW/Position								
	1	2	3	4	5	6	7	8	9
A	CW	CW	CW	CCW		CW	CCW		
B	CW	CW	CCW	CW		CW	CCW		
E	CCW				CW	CCW			
C	CW	CCW				CW	CW	CW	CCW
D	CW	CCW				CW	CW	CCW	CW

Figure 5.3-3: Scatterometer Stick Array Antenna Switching (V-Polarization Shown; H-Polarization Identical)

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CONFIGURATION A4 combines the Microwave Radiometer and Scatterometer into a single 2 meter reflector. It is considered that this approach, even if it could be configured to work, would create too great a risk factor (single point failure). The microwave plumbing and switching to accommodate five frequencies (6.6 to 37 GHz) would be a real "nightmare".

Trade 3 - Phased Array. The Phased Array approach of Configuration A2 has essentially the same problems as the A1 parabolic reflectors.

Trade 4 - Plannar Array. In lieu of the five stick arrays, each 3 meters long, two body mounted 1.5 x 1.5 meter plannar arrays (horizontal and vertical polarization) appear attractive if the sensor could provide a frequency sweep or step to accomplish the scanning. It appears that this approach would have the following advantages:

- o No deployment.
- o No duppler processing (pencil beam).
- o Reduced microwave switching.
- o No articulation.
- o No r.f. moving joints (deployment).
- o Reduced d.c. power requirements.

The one disadvantage would be the broad frequency scan of the sensor transmitters and receivers.

The plannar array evaluation was discontinued after very preliminary consideration due to the understanding that the sensor could not sweep or step frequency.

5.3.3.3 R.F. Switching. The sequencing of the r.f. transmission and reception between five scatterometer antennas for two polarizations at a continuous rate of about ten seconds per cycle creates a serious switching problem. Mechanical devices obviously cannot be considered for the one year life. Solid state type switches, such as circulators or pin diodes, offer the only logical approach to be considered. Circulator switching schemes were considered in the evaluations, although pin diodes would be equally as applicable.

The use of separate transmitters and receivers for the horizontal and vertical polarizations helps to simplify the switching. The recently established requirement for including the calibration antenna at the same sampling rate as the four main antennas has severely complicated the switching scheme. As seen in Figure 5.3-3, nine (9) circulators are required for switching each polarization. Assuming a 0.3 dB insertion loss per circulator, the transmit and receive paths would have a switch loss of up to 1.2 dB.

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If the fifth antenna sampling for calibration could be reduced to once a day, a mechanical switch could be used on that leg. This would reduce the number of circulators required from nine (9) to six (6), as shown in Figure 5.3-4, and the switch insertion loss would be reduced to about 0.9 dB.

It is expected that before the hardware phase, an effort will be made to reduce the scatterometer sensor system to a single transmitter and two receivers. This would expand the transmit/receive modes to V/V, H/H, V/VH, and H/HV which might be an advantage to the data users. Of course, deleting one of the transmitters would further complicate the r.f. switching scheme and increase the insertion loss.

Another switching scheme with only five (5) switches was considered but discarded due to a severe unbalance in the insertion loss to each antenna (0.3 dB to 1.5 dB for the 5 stick, 2 transmitter system). However, the combined transmit/receive loss would be balanced at 1.5 dB. The circuit is shown in Figure 5.3-5.

5.3.4 Tranet Doppler Beacon. The dual frequency quadrifilar helix antenna discussed in the attachment to JPL letter 627-LDA:vr/2.13.4, dated 13 February 1974, is included in the Payload Module baseline. Per JPL request, it will be provided by Boeing; however, if it is operational on the Navy Navigation Satellite System, the minimum cost approach might be for the entire Beacon System (including the antenna) to be GFE. In addition, per JPL Finance request the cost estimate will not be included in the Boeing report but will be provided by JPL.

5.3.4.1 Antenna Deployment. As seen in Figure 5.3-1, three deployment systems are required to deploy the five antennas. The stick arrays are stowed along the payload module and retained by the ordnance operated pin puller(s) during boost. Deployment is accomplished by activating the pin puller and permitting the arrays to pivot until they contact the stops. A simple overcenter latch holds the arrays against the stops and prevents any possibility of backoff during spacecraft maneuvers. The latch is essentially redundant since the actuator preload will hold the array against the stop under all normal conditions.

The deployment system consists of Fabroid pivot bearings, a linear spring and damper actuator assemblies, ordnance operated pin puller(s) and mechanical latch/stop assemblies. The primary deployment components have been flight proven on the Lunar Orbiter and MVM programs and are used for rocket motor deployment on the current Small Satellite contract. The proposed actuator is a highly damped spring thruster that provides a slow rate of deployment to eliminate high impact loads and spacecraft disturbances. Deployment time will be approximately 45 seconds.

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Ant	SW/Position					
	1	2	3	4	5	6
A	CW	CW	CCW	CCW		
B	CW	CCW	CW	CCW		
C	CCW			CW	CW	CCW
D	CCW			CW	CCW	CW

1 Coax Switch Calibration (Ant. E) ≤ 1/Day)

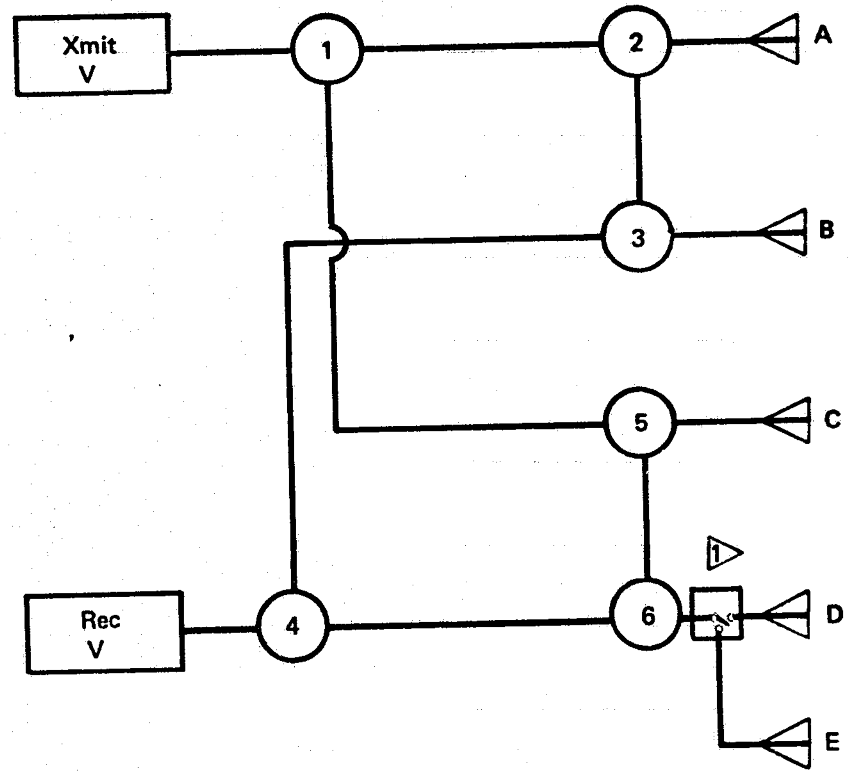


Figure 5.3-4: Scatterometer Stick Array Antenna Switching (V-Polarization Shown; Horiz is Identical)

ANT.	SW/POSITION				
	1	2	3	4	5
A	CW	CCW	CCW	CCW	CCW
B	CCW	CW	CCW	CCW	CCW
E	CCW	CCW	CW	CCW	CCW
C	CCW	CCW	CCW	CW	CCW
D	CCW	CCW	CCW	CCW	CW

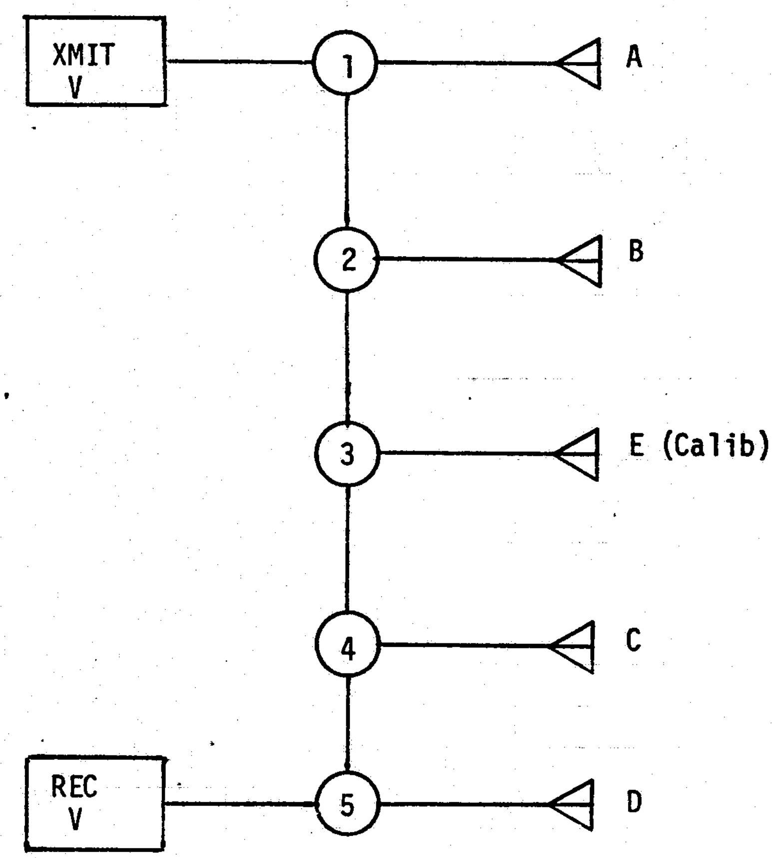


FIGURE 5.3-5: SCATTEROMETER STICK ARRAY ANTENNA SWITCHING
(V-Polarization Shown; H-Polarization Identical)

6.0 SYSTEM TEST

The System Test Program for SEASAT relies heavily on experience gained during similar test programs for Burner II, Burner IIA, SESP 70-1, STP P72-1 satellite and the STP S3 Small Satellites Program.

The test program as shown in Figure 6.0-1 will ensure a thoroughly tested and reliable spacecraft in orbit. The test program offers the following additional features:

- o The spacecraft will be completely functionally checked out, including simulated payload interfaces, prior to mating flight payloads.
- o The test sequence is arranged to provide maximum confidence of meeting all test objectives without impacting scheduled delivery.
- o Field test procedures will be used whenever applicable in spacecraft testing for validation prior to delivery to VAFB.
- o Use of available Boeing-owned automated telemetry ground station receiving, processing and commanding equipment will significantly reduce program costs.
- o Similarity of the Bus Module with the S3 satellite systems and the STP P72-1 systems will make many test techniques and procedures developed during these programs directly applicable to Bus Module testing.
- o All flight spacecraft test functions, will be performed in the contractors facility providing maximum test control and minimum exposure of the spacecraft to transportation and handling environment.

6.1 Development Tests

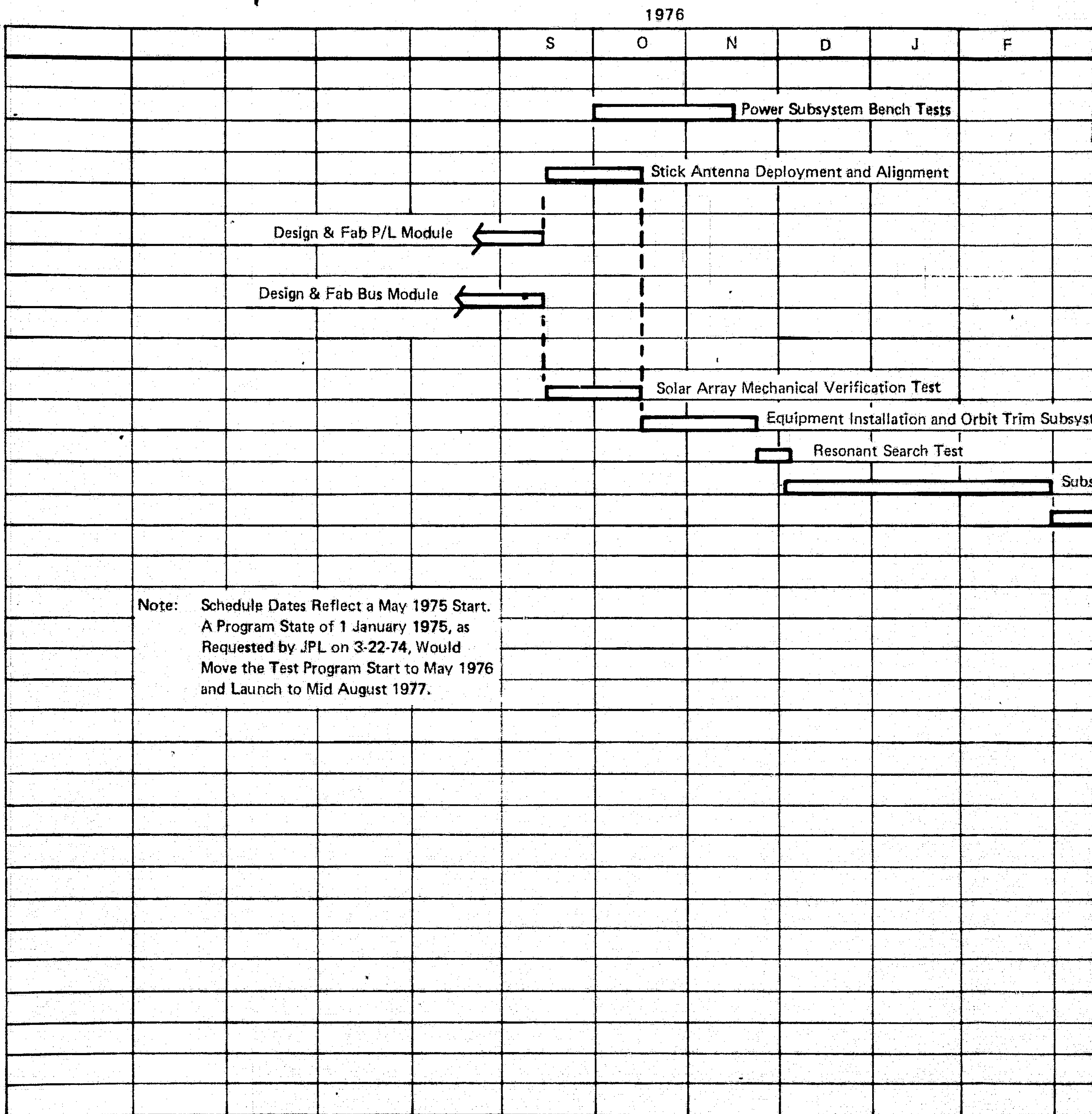
Use of off-the-shelf hardware with little or no modification precludes the need for development testing.

6.2 Type Acceptance (TA) Tests

6.2.1 Components/Subsystems/Assemblies

Type Acceptance (qualification) testing will be performed only on components, subsystems or assemblies which are either new designs or major modifications to existing designs.

Whenever possible, off-the-shelf components will be selected with previous qualifications for environments equal to, or exceeding, those required for this program. Procured components not previously qualified will be tested by the vendor in accordance with a Boeing procurement specification reflecting the requirements defined in the RFQ. Boeing and government source inspection will be included as applicable to ensure compliance with specification test requirements.


1976
S
O
N
D
J
F
Power Subsystem Bench Tests
Stick Antenna Deployment and Alignment
Design & Fab P/L Module
Design & Fab Bus Module
Solar Array Mechanical Verification Test
Equipment Installation and Orbit Trim Subsyst
Resonant Search Test
Subs
Note: Schedule Dates Reflect a May 1975 Start. A Program State of 1 January 1975, as Requested by JPL on 3-22-74, Would Move the Test Program Start to May 1976 and Launch to Mid August 1977.

FOLDOUT FRAME 2

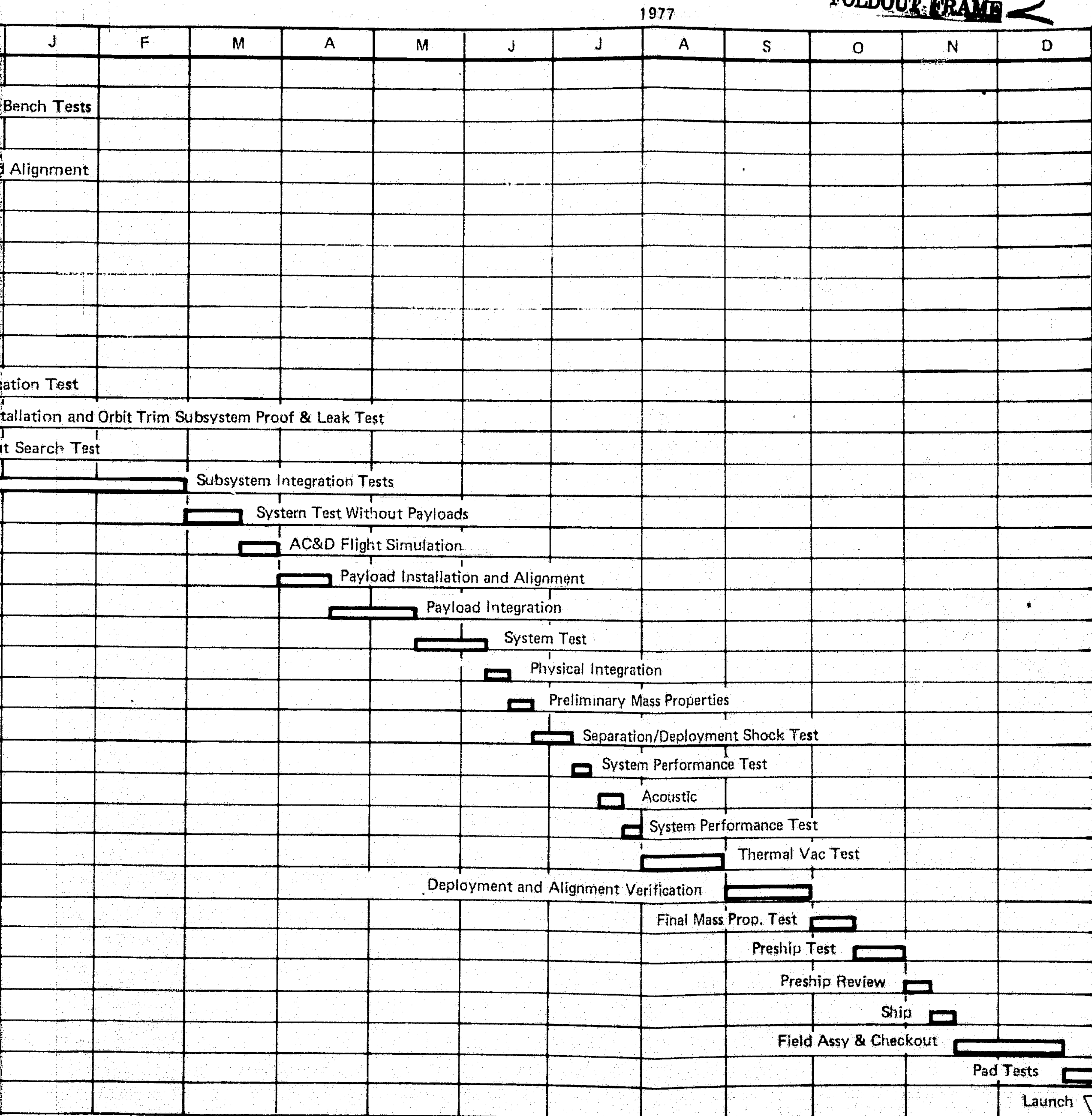


Figure 6.0-1: SEASAT Test Schedule

Where TA tests are required, the tests will be performed on the first set of hardware. This hardware will become the flight spare unit and will be used for flight in the event that a catastrophic failure occurs to the flight hardware. This hardware will be used for a second spacecraft launch in the event of a catostrophic failure of the first spacecraft after launch.

The TA testing will consist of exposing the hardware to sufficiently higher than expected launch flight environment levels to demonstrate adequate margin while operating simulated launch/flight modes. Performance will be verified prior to, during, and after exposure to the environments.

6.2.2 System

No TA tests are planned at the spacecraft level. Confidence in the design will be obtained as a result of:

a. Qualification by test or similarity of the components/subsystems/assemblies comprising the spacecraft.

b. Performing the spacecraft level flight acceptance testing at environmental levels which represent expected flight levels.

6.3 Flight Acceptance (FA) Tests

6.3.1 Components/Assemblies

Flight Acceptance Testing will be performed on flight components or assemblies and will consist of exposing the hardware to slightly higher than expected launch/flight environments while operating in simulated launch/flight modes. Performance will be verified prior to,during, and after exposure to the environments.

6.3.2 Bus Module Tests

The following tests as shown on figure 6.0-1,except for the Power Subsystem Test, will be performed on the Bus Module. The Power Subsystem Test will be performed on the bench prior to installation of the subsystem in the Bus Module.

6.3.2.1 Power Subsystem Bench Test

The power subsystem components and simulated spacecraft wiring are first interconnected and operated as a subsystem on the bench. Operation during most of this testing will utilize laboratory power supplies with a voltage/current output characteristics similar to that of a solar array; one test battery set is utilized during the bench and subsequent tests to verify operation of the subsystem charge controllers and boost regulators.

During the bench test period clear understanding of operation under dynamic load changes, commanded mode changes and failure conditions is determined. Where incompatibilities are found corrective action can be taken with minimum program impact. Tests will be performed and measurements taken

to establish noise and ripple levels, and transient response characteristics to provide a baseline for evaluating overall spacecraft performance during integration and system performance tests.

6.3.2.2 Solar Array Mechanical Verification Test

Deployment testing of the solar array structural and mechanical subsystems will be conducted to verify functional characteristics of the release system, deployment springs, dampers, and latches and to insure that critical deployment margins and alignment tolerances are met. The test configuration will consist of flight or flight equivalent structure and mechanisms and mass simulated electrical components. Flight equivalent wire bundles and electrical cable will be installed across movable joints to simulate their effect on deployment characteristics. The test configuration will be mounted to a test fixture that simulates module interfaces, supported by a zero g suspension system, and deployed several times to insure data repeatability. Deployment forces, friction loads, alignment repeatability, and structural stiffness will be measured. Movies of the deployment sequence will be taken.

6.3.2.3 Subsystem Integration Tests

Prior to application of power to the spacecraft, open-circuit voltage-no-voltage tests will be made on all spacecraft connectors, to verify the presence of the correct voltage at the proper pins on each connector, and absence of voltage on all other pins. After the open-circuit checks are complete the loads will be connected, power applied, and voltage and current measurements made at the spacecraft loads.

Subsystem integration tests will be performed: (1) to minimize the probability of equipment damage due to improper interfaces; (2) to test all interfaces between subsystems and verify that their signal characteristics agree with the circuit data sheets: (3) to evaluate the performance of each subsystem while it is operating on spacecraft power and in the spacecraft environment; and (4) to provide a preliminary evaluation of subsystem effects on the system environments. Interfaces will be verified systematically. The sequence of testing will vary with hardware availability. The subsystems will be exercised open circuit to verify the voltage-no-voltage characteristics at each interfacing connector prior to making any connection. After interface verification, the subsystems will be tested as follows:

Power Subsystem - The power subsystem will have been completely verified during the bench level test discussed in section 6.3.2.1 Additional testing will be performed to verify its performance and compatibility with the satellite wire bundle. The power subsystem will then be utilized with a solar array simulator to power the spacecraft during the remaining tests.

TT&C Subsystem - Powered from the S/C Bus Power Subsystem, the TT&C will be checked out for those system level performance characteristics which can be satisfactorily measured without all other subsystems connected. Included would be those elements involving subcarrier, composite telemetry signal, carrier modulation indices, r.f. power, uplink threshold, command decoding and storage. Upon the connection of other subsystems, the telemetry processing and command execution functions would be checked out.

Throughout System Level testing the Boeing Test Data Van will be used to test and monitor performance. All TT&C functions can be adequately checked out except for Ranging and phase and group delay. It is recommended that a Ranging Test Set be provided as GFE for use at the AFWTR or Kent to allow these verifications.

AC&D Subsystem - AC&D performance will be monitored through test connectors provided on the FCE, programmer and command/control unit leaving interfaces between the various satellite subsystems intact and minimizing the addition of EMI sources. Verification of functional integrity will be accomplished by real-time and stored commands commanding various modes and the resultant control action monitored as magnetic field intensity variation at the magnetic torquing coils. Test magnetometers will provide a go-no-go indication of field strength and polarity changes. Verification of momentum wheel speed control will be accomplished by injecting pitch attitude errors and monitoring the momentum wheel tachometer signal. Horizon and sun sensor signals through telemetry interfaces will be verified by use of sensor stimulators. The magnetometer will be verified by stimulating the test coils that are an integral part of the sensor unit.

After the completion of the above subsystem tests, telemetry calibration will be performed to verify that the stability and accuracy of each telemetry channel. This will be accomplished by inserting calibrated voltages into the spacecraft interface connectors and verifying the values received from the Telemetry system at the Telemetry ground station.

6.3.2.4 System Test without Payloads

After the subsystem integration tests have been completed, a bus module system test utilizing payload electrical simulators or breakout boxes will be performed. This test will verify readiness of the Bus Module for payload installation and integration. The Module will be subjected to simulated prelaunch, boost and orbital sequences to verify proper operation of the primary, secondary, redundant and backup modes.

The test will verify proper operation of the power subsystem using simulated solar array power in conjunction with all bus module subsystems. Module commands will be transmitted and proper module response will be verified through the telemetry system. Module housekeeping status information and AC&D outputs will also be verified through the telemetry system with sensor stimulation as required.

6.3.2.5 AC&D Flight Simulation Test

The Attitude Control and Determination simulation test provides a closed loop dynamic system level test to verify compatibility between AC&D system elements and other bus subsystems including wiring, power, orbit trim and TT&C. The test will be conducted after completion of all Bus subsystem integration tests. Analog computers will be used to simulate the mass properties of the satellite by scaled integrations of momentum wheel drive and magnetic torques signals from bus test connectors. The computed vehicle attitude will be used to modify horizon sensor attitude signals to provide closed loop operation that verifies dynamic performance and compatibility with the integrated bus environment including signal noise.

A special run will be made during the test to generate real-time and playback telemetry tapes for verification of attitude determination software compatibility and integration in the total SEASAT program software.

6.3.3 Payload Module Test

6.3.3.1 Stick Antenna Deployment and Alignment

Deployment testing of the stick antenna and solar array structural and mechanical subsystems will be conducted to verify functional characteristics of the release system, deployment springs, dampers, and latches and to insure that critical deployment margins and alignment tolerances are met. The test configuration will consist of flight or flight equivalent structure and mechanisms and mass simulated electrical components. Flight equivalent wave guide will be installed across movable joints to simulate its effect on deployment characteristics. The test configuration will be mounted to a test fixture that simulates module interfaces, supported by a zero g suspension system, and deployed several times to insure data repeatability. Deployment forces, friction loads, alignment repeatability, and structural stiffness will be measured. Movies of the deployment sequence will be taken.

6.3.4 Spacecraft Tests

6.3.4.1 Orbit Trim Subsystem Proof and Leak Test

After mating the bus and payload modules the Orbit Trim Subsystem components will be installed and the spacecraft will be installed in a proof test cell. The subsystem will be pressurized and the appropriate proof and leak tests performed. The proof test will be performed on the system at a level of 1 ½ times the operating pressure. All joints and possible leak points in the system will be leak checked at operating pressure.

6.3.4.2 Resonant Search Test

A resonant search test of the total stack above Atlas station 502 will be conducted to verify the stiffness model and insure that minimum cantilever frequency requirements are met. The test configuration will consist of the flight structure for the payload module, bus module, Burner II stage, and adapter system. Actual or mass simulated equipment and payload components will be installed to provide a flight equivalent weight. The test configuration will be mounted to a rigid base fixture and subjected to a low level sine vibration sweep from approximately 4 to 30 Hz in two lateral axes. Electro-magnetic vibrators will be used to excite the test configuration and accelerometers used to establish mode shapes at major resonances. Input force levels will be controlled to limit maximum responses to 1.0g. The Burner II stage and adapter structure are included in the test configuration to provide a more meaningful test and to allow the use of existing test fixturing.

6.3.4.3 Payload Integration

After all Bus Module tests have been completed utilizing payload simulators or breakout boxes, the flight payloads will be connected to the spacecraft wire bundles and each payload will be exercised through all modes of operation by the spacecraft as commanded from the Boeing Test Data Van. The payload data channels will be transmitted through the spacecraft telemetry system and monitored, recorded and verified at the Test Data Van.

The recorded information on magnetic tape will be available to the payload contractors for further analysis if necessary.

Prior to the start of this test the characteristics of each signal going to the payloads from the spacecraft will have been verified. During the test the characteristics of each signal going to the payload and from the payload will be verified by utilizing a breakin box between the payload and the spacecraft. This test will verify there are no marginal signals or transients in the systems which could degrade the orbital performance of the spacecraft.

6.3.4.4 System Test

System test will be performed to verify the performance of the spacecraft as an integrated system within the constraints of the support equipment and the earth environment. The spacecraft will be exercised through simulated flight sequences of operation to verify proper operation in primary, secondary, redundant and backup spacecraft modes.

The initial system test will include a power profile and transient test to determine the power dissipation of each spacecraft subsystem in all pertinent steady state modes of operation, turn-on transients, and heater resistance. Subsystems will be brought to launch mode condition and exercised through each significant electrical state to determine power dissipation.

Subsystems and payloads will be operated in typical orbiting sequences to verify that performance is not degraded as a result of subsystem/payload turn on/off, sequence timing and/or simultaneous operation.

During this test, magnetic tape recordings of the spacecraft data will be made. These tapes will be provided to the customer for use by the STDN ground station network to verify spacecraft compatibility prior to receipt of the spacecraft at Vandenberg Air Force Base.

6.3.4.5 Physical Integration

Tests will be performed to verify the spacecraft handling procedures and to verify the compatibility of the handling equipment with the spacecraft hardware. Additional tests will be performed to verify compatibility of the interfaces between the launch vehicle and spacecraft. Clearance between the spacecraft and launch vehicle shroud will also be verified.

6.3.4.6 Mass Properties Test

Mass property tests will be performed to statically and dynamically balance the total spacecraft including the Burner II stage and to verify that design weight, center of gravity, and inertia requirements are met. The bus module, payload module, and Burner II stage minus the rocket motor will be in the ready for flight configuration. The rocket motor will be weighed and balanced at the vendor facility and shipped directly to the field. Testing will be conducted on the individual modules as well as the total spacecraft to verify all critical mass property parameters. Specific tests include spin balancing of the modules, Burner II stage, and total system, moment of inertia measurement of the combined bus and payload modules, and weighing of the individual modules and Burner II stage.

The weight and balance procedures and all test results will be recorded in a Weight and Balance Handbook. The handbook will include detailed procedures for maintaining strict mass property control from final testing until launch.

6.3.4.7 Separation/Deployment Shock Test

The complete spacecraft system will be subjected to actual separation and deployment pyrotechnic shocks during acceptance testing to detect possible workmanship errors and faulty parts. The payload module, bus module, and Burner II stage will be assembled using flight equivalent separation and deployment ordnance. A counterbalance system will be used to simulate zero "g" and to permit separation at the separation planes. All separation and deployment ordnance will be sequentially fired one time. A System Performance Test will then be conducted to verify no functional degradation.

6.3.4.8 System Performance Test (SPT)

A System Performance Test (SPT) will be performed prior to and after each environmental test. These performance tests will verify the proper functional operation of the spacecraft subsystems and payloads. The tests will be automated utilizing the Boeing Test Data Van and will be similar to, but an abbreviated form of, the System Test described in Section 6.3.4.4.

6.3.4.9 Acoustic Test

The complete spacecraft system will be subjected to the maximum expected flight acoustic environment for one minute to detect possible workmanship errors and faulty parts. The bus and payload module and Burner II stage will be assembled on a test fixture and installed in the acoustic test chamber for the test. A system performance test will be performed following the test to verify no functional degradation.

6.3.4.10 Thermal Vacuum Test

The spacecraft system thermal vacuum test will provide an operational performance verification and qualification of a near flight configuration in a simulated orbital environment. Satellite subsystem and experiments

will be operated through all planned orbital and backup modes. Test data will be incorporated into the thermal control computer math model used to predict flight temperatures. The test will be performed in Chamber A at the Boeing Kent Space Center, Kent, Washington.

Test Configuration. The test configuration will include the payload and bus module in a flight configuration except as noted below.

- o Solar array power will be simulated
- o Partial dummy solar arrays will be installed and articulated
- o Stick antennas will not be installed.
- o The N_2H_4 system will not be charged.

Test Modes. Two steady state tests will be performed to qualify the satellite under extreme hot and cold conditions. Internal electrical loads and the external environment will be adjusted to the appropriate worst hot or cold case. Following each of the steady state tests the satellite will be operated for ten orbits under fully operational conditions (except as noted above) followed by five orbits at a reduced level of operation. All testing will be performed at chamber pressures of 10^{-5} torr or less and with the chamber shroud temperature at -290°F or less.

Test Setup. The satellite will be mounted to a test fixture at the three separation bolt holes on the aft end of the bus module and by a column support placed beneath the payload module. The test fixture and satellite will be static except for spacecraft articulated systems. During steady state and the transient Beta =0° simulated orbits the environment will be simulated with arrays of infrared quartz lamps and infrared panels. The temperatures and thermal heating fluxes will be programmed to match real time orbital variations including sunset/sunrise. During the Beta = 90° transient simulation the front side infrared sources will be partially replaced with a local Xenon arc solar simulator beam. This source will emanate from the chamber lid and will be folded onto the solar absorber panel with a flat mirror. In addition, elements of simulated solar energy may be directed onto the parabolic antennas to simulate worst case temperature distortions.

A summary of test times and conditions are shown in Table 6.3.4.10-1.

Instrumentation. Communication and telemetry will be hardlined through the chamber walls. Between 50 and 100 temperature sensing channels will be provided. The data will include real-time television presentation of up to forty channels, a real-time computer printout of all channels and subsequently 8 ½ X 11 time-temperature plots of all desired channels.

TABLE 6.3.4.10-1 SUMMARY OF THERMAL VACUUM TESTING

TEST	TYPE	PURPOSE	TIME (HOURS)
HOT	STEADY STATE	QUALIFY WORST HOT CASE	10
NOMINAL ($\beta=0^0$)	TRANSIENT	VERIFY NOMINAL PERFORMANCE	17
REDUCED OPERATION ($\beta=0^0$)	TRANSIENT	VERIFY REDUCED TIMELINE PERFORMANCE	8.5
COLD	STEADY STATE	QUALIFY WORST COLD CASE	10
NOMINAL ($\beta=90^0$)	TRANSIENT	VERIFY NOMINAL PERFORMANCE	17
REDUCED OPERATION ($\beta=90^0$)	TRANSIENT	VERIFY REDUCED TIMELINE PERFORMANCE	8.5

6.3.4.11 Antenna and Solar Array Final Deployment and Alignment

The flight antennas and solar arrays will be aligned and deployed during system level acceptance tests to verify deployment margins, alignment, and performance of release mechanisms and latches. The test configuration will include the flight modules with complete antenna and solar array systems. Zero g will be simulated by a spring/counterbalance system. The antennas and solar array will be correctly aligned in the deployed position and a number of deployment cycles conducted to verify critical performance parameters. Deployment forces, friction, alignment, and deployment times will be measured, recorded, and adjusted where necessary to meet established requirements. Following the tests the antennas and solar arrays will be carefully stowed in the final configuration for launch.

6.3.4.12 Preshipment Test

Prior to shipment of the spacecraft to Vandenberg Air Force Base for field processing, the spacecraft will be subjected to a System Test as described in Section 6.3.4.3. Prior to shipment a review of all test and Quality Control records will be performed to verify readiness of the spacecraft for shipment to VAFB.

6.4 Field Operations

The primary purpose of spacecraft operations at VAFB is to prepare the spacecraft for launch. This activity includes an evaluation of spacecraft performance after shipment and demonstration of spacecraft compatibility with the STDN ground stations, the launch complex, and the launch vehicle.

6.4.1 Spacecraft Assembly and System Test

The spacecraft will be assembled in a system test configuration and functional tests similar to those conducted at Kent will be performed. These tests will utilize the same AGE which was used during Kent testing. During these tests the payload sensors will be stimulated to verify operational readiness of the payloads.

After completion of system testing, spacecraft compatibility with the appropriate STDN ground stations will be verified via hardline of RF links. Proper launch configuration of the antennas and solar arrays will be verified by visual inspection.

The satellite will then be prepared for transportation to the launch pad for installation on the launch vehicle.

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6.4.2 On Pad Tests

After installation on the launch vehicle and prior to installation of the shroud, a short confidence test of the spacecraft will be performed. This test will include payload stimulation and checkout where feasible. The test will be performed utilizing the Boeing owned Test Data Van located adjacent to the pad. During this test and until shroud installation, the spacecraft will be protected by a portable environmental enclosure installed in the launch pad servicing tower. The enclosure will maintain the spacecraft environment within acceptable temperature, humidity and cleanliness limits.

7.0 AEROSPACE GROUND EQUIPMENT (AGE)GROUND HANDLING EQUIPMENT (GHE)

The following AGE and GHE will be required to process, test and ship the spacecraft at Kent and at VAFB.

7.1 AGE

7.1.1 Contractor Furnished Electrical/Electronic Ground Equipment

A Boeing owned Test Data Van (TDV) will be used to monitor the spacecraft telemetry and control and spacecraft during all test operations. The TDV utilizes a mini computer and an Aerospace command Encoder to command the satellite and automatically display, print and verify spacecraft limits for selected format locations from the telemetry bit stream. The TDV also records all the telemetry information on magnetic tape for post test evaluation if necessary or as required by the payload contractor personnel for their evaluation.

Boeing will also provide the test support equipment necessary to test the spacecraft. This will include such items as: breakout boxes; spacecraft sensor stimulators; battery servicing equipment; cabling; antenna hoods; a squib simulator test set; and an electrical test set to permit hardline monitoring and control of spacecraft functions (such as power application and timer stepping). Boeing will also furnish all the standard test equipment required, such as, oscilloscopes and meters as capital equipment items.

7.1.2 Government Furnished Electrical/Electronic Ground Equipment

It is expected that equipment necessary to verify performance of the payload instruments while installed on the spacecraft will be provided as GFE by the payload contractors. Included are such items as, r-f measurement equipment and instrument stimuli. Payload electrical simulators to be used in system checkout prior to the installation of the payloads are also expected to be GFE.

7.2 GHE

7.2.1 Contractor Furnished

Boeing will provide the necessary spacecraft shipping containers, handling slings, dollies, protective covers, and test fixtures necessary to process the spacecraft in Kent and at VAFB. Boeing will also furnish a portable launch pad servicing tower environmental enclosure to provide spacecraft protection prior to shroud installation. The enclosure will utilize the existing air conditioning available in the servicing tower or from the umbilical mast.

7.2.2 Government Furnished GHE

It is expected that any unique handling or servicing equipment required by the payloads will be provided as GFE. It is also assumed that a N_2H_4 Servicing cart will be made available to the program from an existing GFE source.

8.0 PROGRAM PLANS

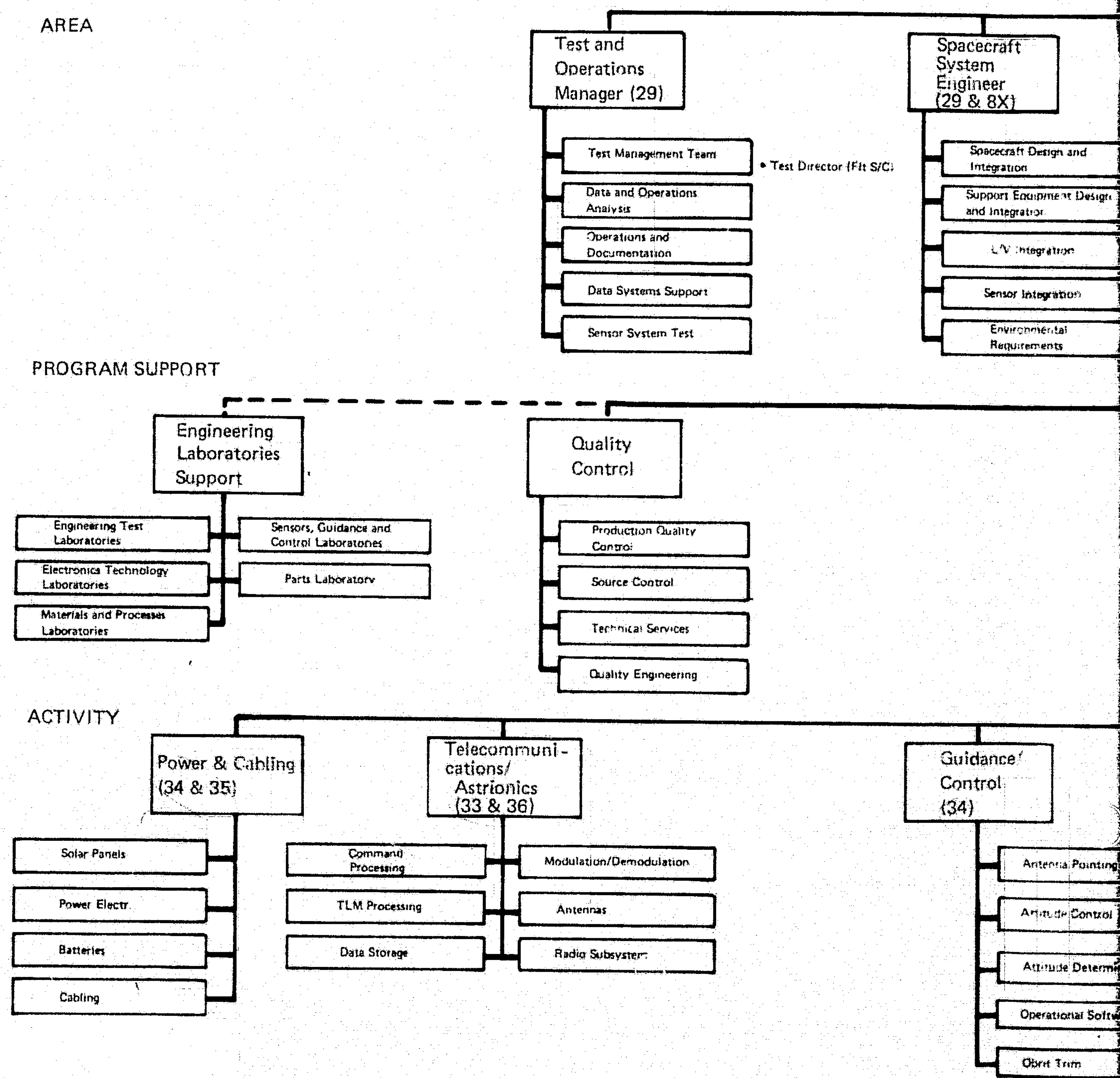
This section briefly covers the Boeing Company initial program planning for accomplishment of the SEASAT mission. The planning reflects recognition of the following elements which are key to successful accomplishment of program technical objectives within costs and on schedule.

- Spacecraft design requirements are dictated by payload, mission operation, data and tracking, and launch vehicle requirements. Planned interface activity, reflecting experience on previous programs, provides assurance requirements are understood, incorporated and verified.

- Reliability and Quality Assurance are continuous militant functions from Contract Go-ahead until launch of the spacecraft. Reporting directly to the Program Manager, these functions are assured proper program emphasis.

- Program schedules reflect realistic hardware lead times, recognize the probability of unforeseen problems, and recognize the relationship between software and hardware availability.

- Key to controlling the technical, schedule, cost and SEASAT interface activities are the activity managers. Figure 8.0-1 depicts the organization, organization activities and area responsibilities that interface with their JPL counterparts. The organization provides for a reasonable span of control in each activity area and is planned on the basis that each activity manager is in essence a subprogram manager within his specialty area reporting to the program manager. The activity manager controls costs, schedules, changes, and technical integrity for his entire subsystem including supplier costs, Quality Assurance and Manufacturing costs and schedules.



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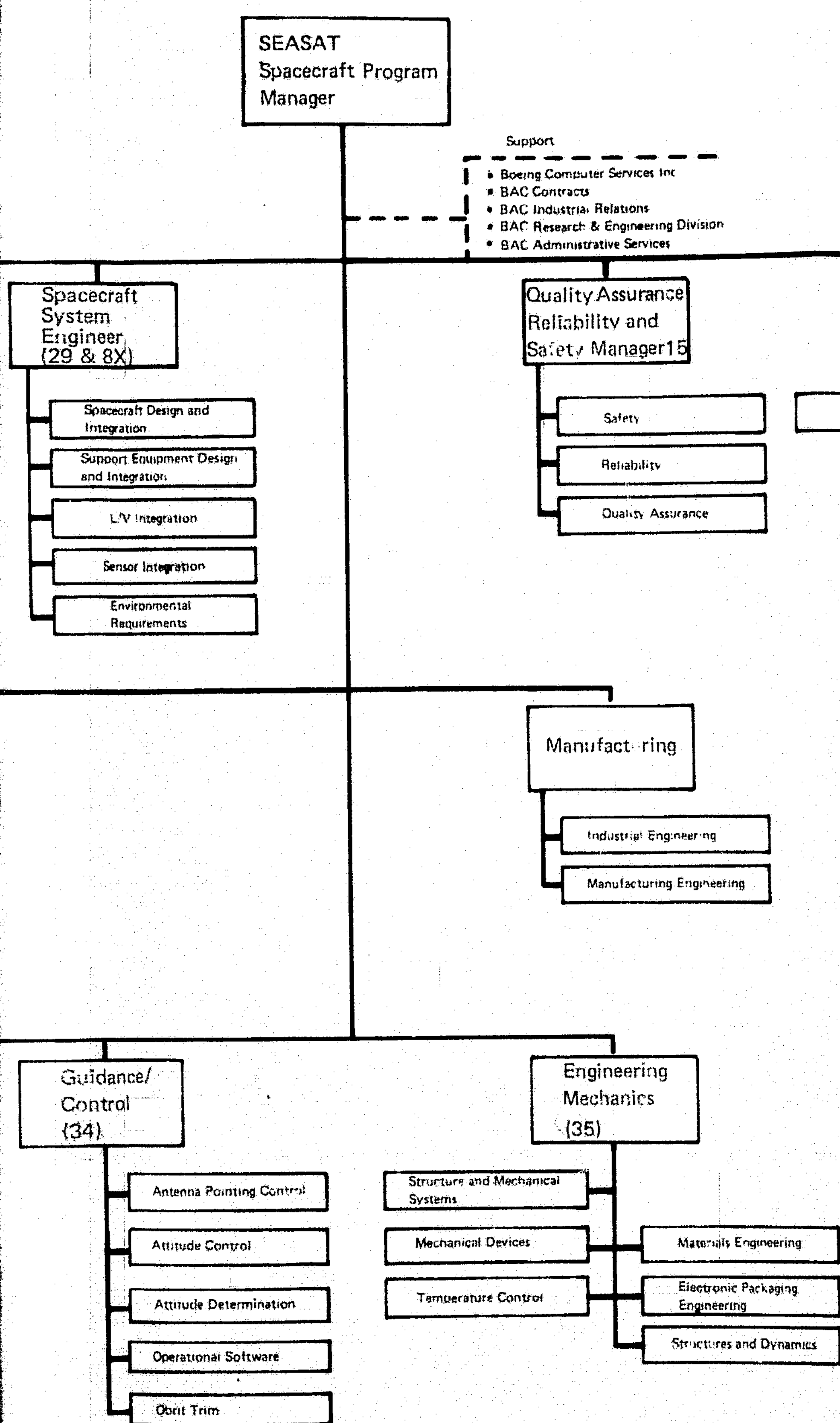

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Figure 8.0-1: SEASAT-A Organization

8.1 SCHEDULE

The Master Schedule, Figure 8.1-1 is based on a specified SEASAT launch early 1978. This schedule represents a total program flow. An earlier start date, as requested by JPL, of January 1975 rather than May 1975 would move the launch date back 4½ months to August 1977, with no change to program phasing or length.

The cirtical path is the procurement bar with 12-16 months lead time on major procured items. The total schedule flow has considered some time for problems in test and is a schedule that can be achieved using a normal work week (five 8-hour days) through the life of the program. At the time of program definitization, detail schedule segments are prepared at the subsystem levels down to component identification and schedules. Shortage reports are developed and reported from these schedules. Purchased equipment is followed at the piece part, sub-assembly, assembly, and test levels at suppliers' plants and is monitored by periodic audits in addition to supplier formal reporting. Internal and external schedules are monitored weekly in the program manager's meeting.

SEASAT MASTER SCHEDULE

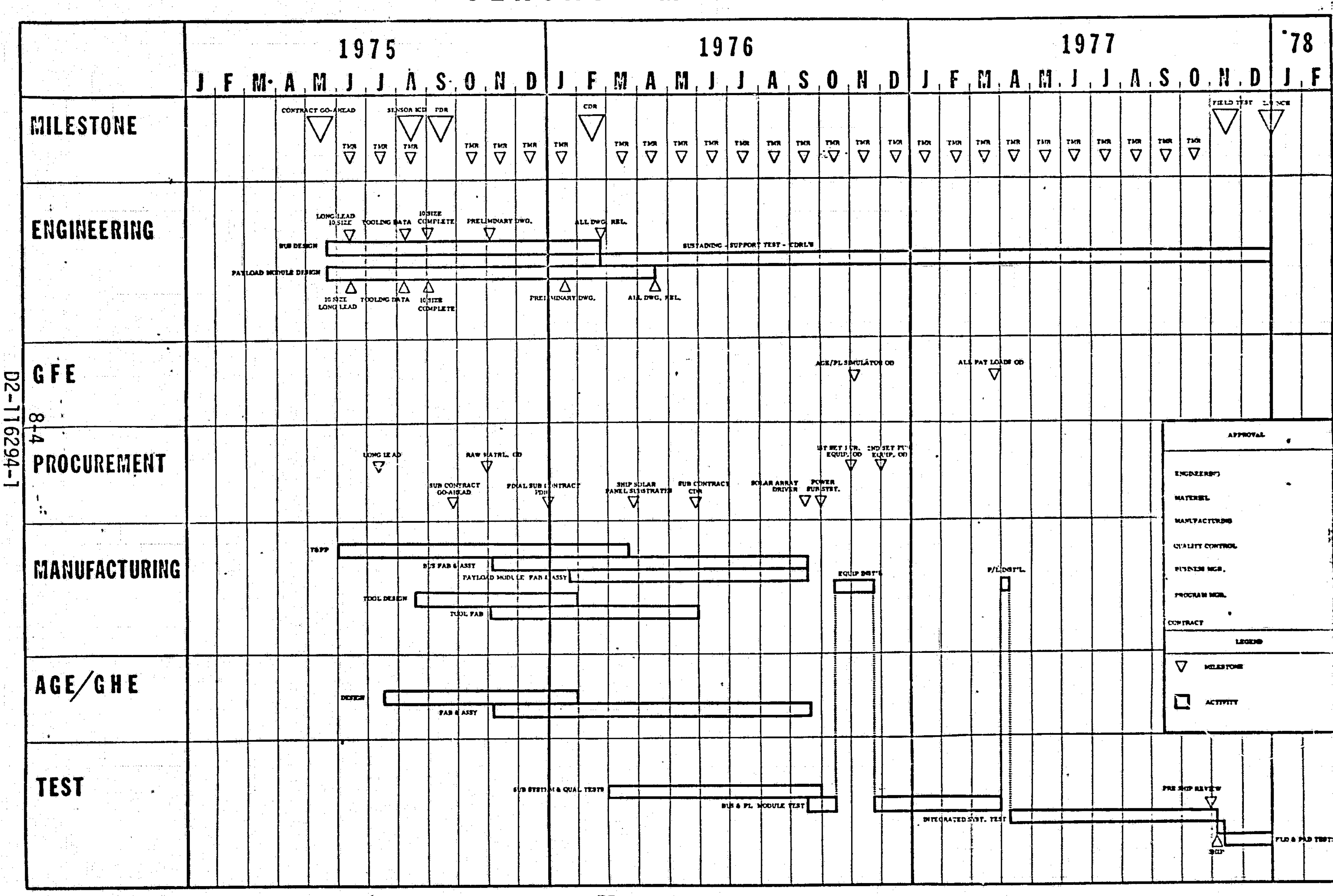


FIGURE 8.1-1

8.2 INTEGRATION

The integration process, depicted in Figure 8.2-1, applies proven integration plans and procedures throughout the definition, acquisition, test and launch phases to define, coordinate, and validate all spacecraft elements of hardware, software, data, facilities, and organization interfaces required to support the acquisition of SEASAT-A useful mission data.

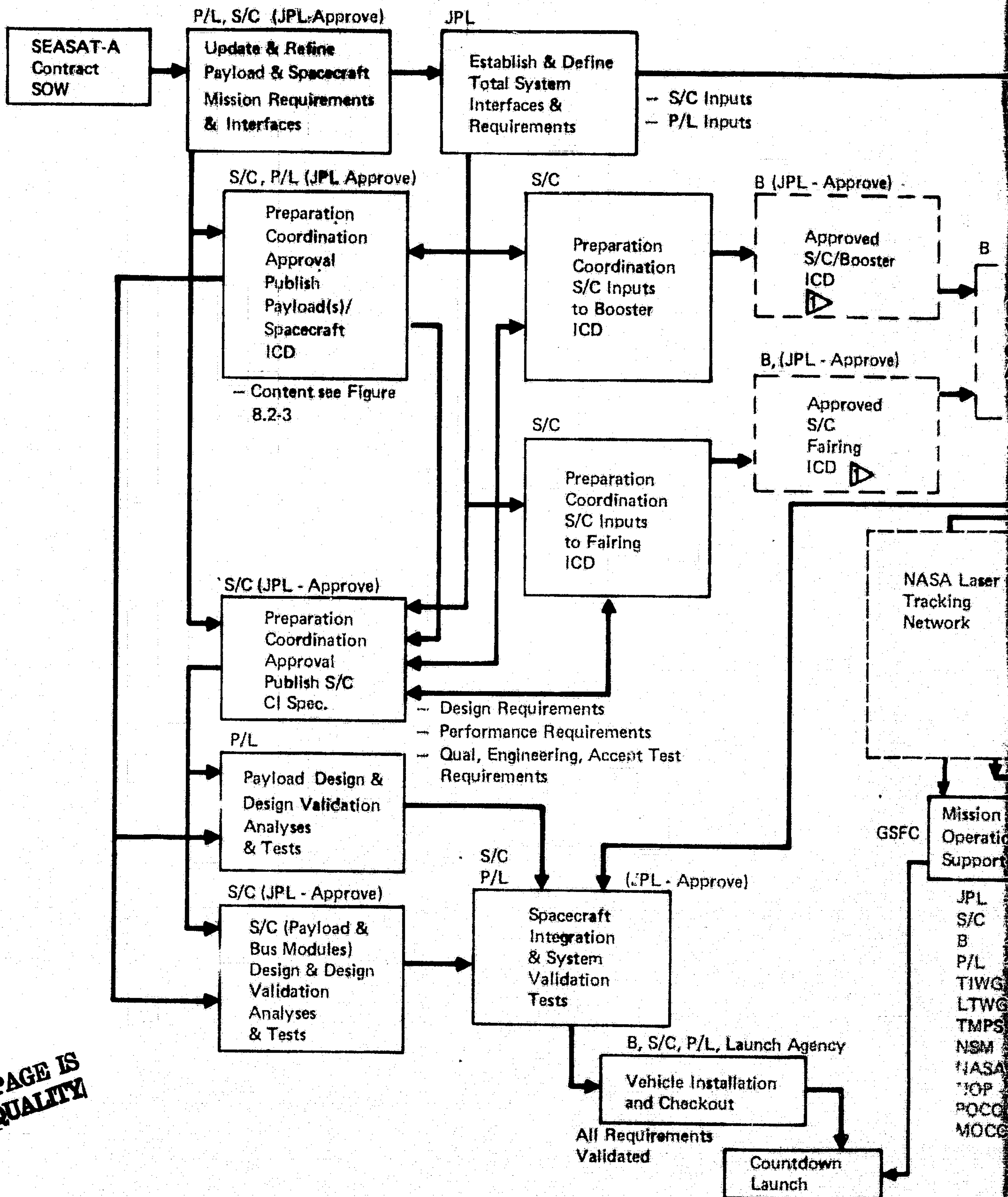
Interface control and integration of the spacecraft (consisting of a Payload Module and a Bus Module) with the Payload(s) element, and integration and interface definition of the spacecraft element with the Booster element and with the Adapter and Fairing elements will be accomplished through ICD/Supplementary Documents. These documents define and control the approved physical, functional, electrical, and procedure/data interfaces including operations. Payload physical and functional interfaces are continuously refined via aggressive coordination with the payload contractors, and finalized to provide early identification and control of design interfaces in an approved ICD. Preliminary ICD(s) are to be available for the preliminary design review (PDR).

The Satellite System Integration Plan (SIP) schedules all exchanges of spacecraft hardware, software, data, facilities, and services as well as the tasks and responsibilities of the Satellite System Contractor with each of the SEASAT participating organizations shown in Figure 8.2-1.

Detailed coordination of plans, procedures, and activities is accomplished at working group meetings. This provides for:

- o Close cooperation with the payload contractor and knowledge of his hardware and requirements to minimize interface design, test, and operational problems.
- o Early freezing of necessary design interfaces in ICD to prevent cost and schedule impact but retaining flexibility to work details of interfaces.

All coordination between Spacecraft System Contractor and other participating elements of the SEASAT program formally flows through JPL with the Sensor Agencies, Launch Vehicle Agency, Launch Agency, Mission Operations and Control Agency, and the Tracking and Data Agency to accomplish the spacecraft interface definition and integration. Satellite System Contractor recognizes the need and will perform informal coordination/clarification with those elements of the SEASAT program organization that have a spacecraft interface. The major tasks are identified on Figure 8.2-1.



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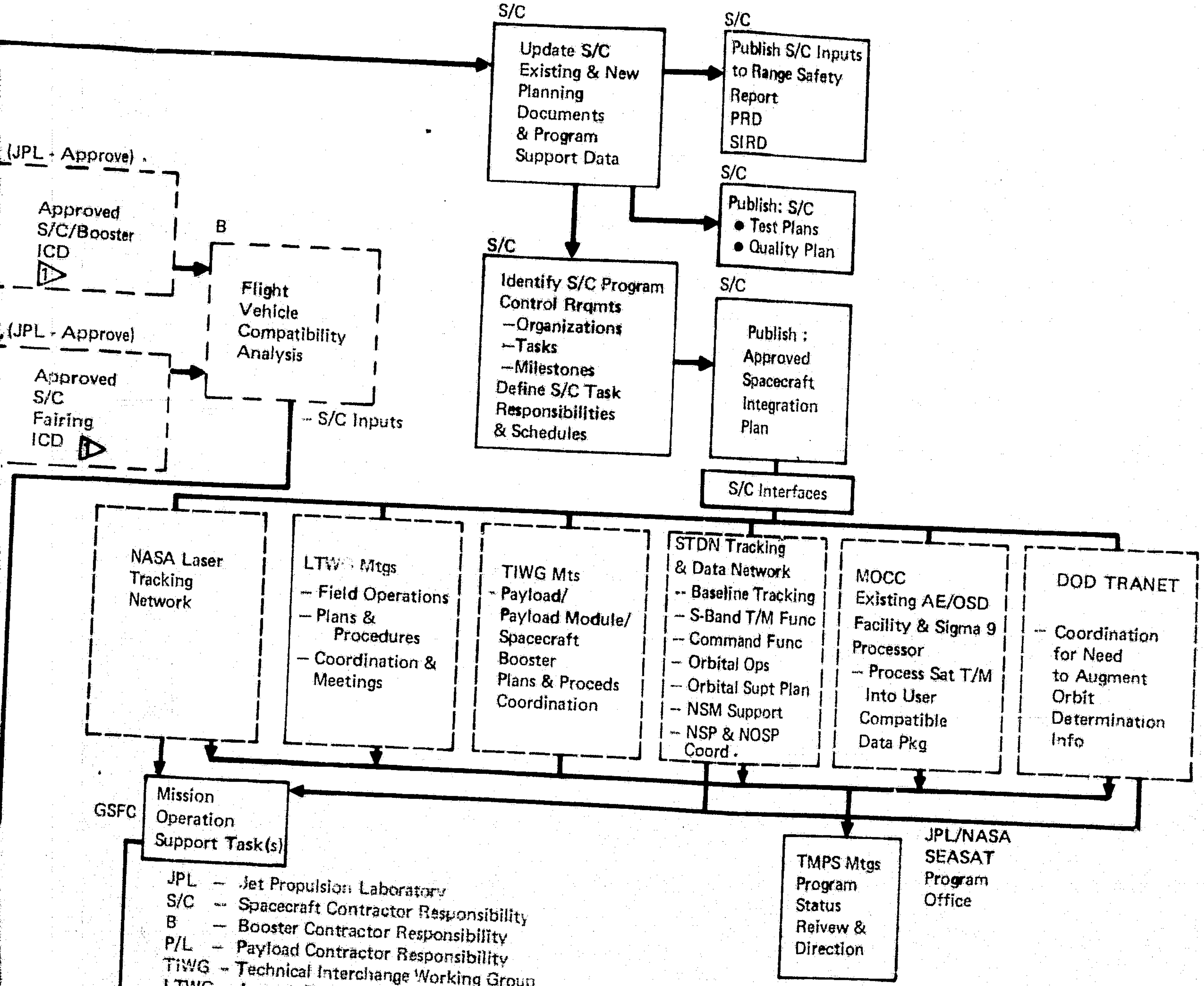

JPL – Jet Propulsion Laboratory
S/C – Spacecraft Contractor Responsibility
B – Booster Contractor Responsibility
P/L – Payload Contractor Responsibility
TIWG – Technical Interchange Working Group
LTWG – Launch Test Working Group
TMPS – Technical Management & Program Status
NSM – Network Support Manager
NASA – National Aeronautics & Space Administration
NOP – Network Operations Plan
POCC – SEASAT-A Project Operations Control Center
MOCC – Mission Operations Control Center
(AE/OSO Facility & Sigma-9 Processor)
(Stdn Tracking & Data Acquisition)
(NASA Laser Tracking Network

TRANET – DOD Tracking and Data Acquisition Network
DOD – Department of Defense
CI – Configuration Item Specification
GSFC – Goddard Space Flight Center
(---) – Dashed Lines Denote Other SEASAT Supporting Agencies
1 Proposed Baseline Atlas F Booster Interfaces Are Established and Existing. For Alternate Delta Booster Interfaces Need to be Developed

Figure 8.2-1: SEASAT-A Integration Process

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8.2.1 Program Organization Interface. Boeing with JPL cognizance will interface with all participating organizations as required to plan and monitor the overall spacecraft activities. The Spacecraft Contractors organizational interface with the SEASAT-A program is depicted in Figure 8.2-2. Thusly, Contractor interprets JPL interfaces as follows:

- o Monthly management reviews to assess the program status; prime areas of status are technical, schedule, and costs.
- o Informal technical coordination at subsystem level (records of coordination, e.g., telecons, meetings minutes, to be maintained and submitted to JPL program office).
- o JPL to approve all interface control documentation, i.e, Spacecraft with sensor interfaces, booster interfaces, and STDN.
- o JPL to support all interface meetings.
- o JPL attendance at subcontractor design reviews, etc. at JPL option.
- o JPL to review and approve system level test plans and requirements.
- o JPL representatives on site during system level tests.

Organization interfaces with other Program Agencies is interpreted as follows:

- o Each Agency is considered as Associate Contractor for technical coordination. Development of interface controls and agreements and interface verification are normal activities.
- o All Program Agency related activities conducted by Boeing under cognizance and contractual direction of JPL.

8.2.2 Payload Interfaces. The Boeing Spacecraft configuration is a satellite totally integrated around the SEASAT-A Payload antennas (altimeter, radiometer, and scatterometer), associated GFE, RF, and electronic components listed in 5.1 and optimized for physical and thermal interfaces within launch vehicle packaging constraints. All payload interface requirements, thermal, structural, power, TT&C, cleanliness, etc., reference Figure 8.2-3, will be met with the proposed modularized spacecraft design. System analysis functional flows and schematic diagrams serve as basis for identification of payload interfaces.

The ICD documents, (typical required content as defined in Figure 8.2-3,) shall record the interface agreements between the two contractors. These documents shall provide the means to identify, evaluate and control all interdependent interfacing design parameters between the payloads and the

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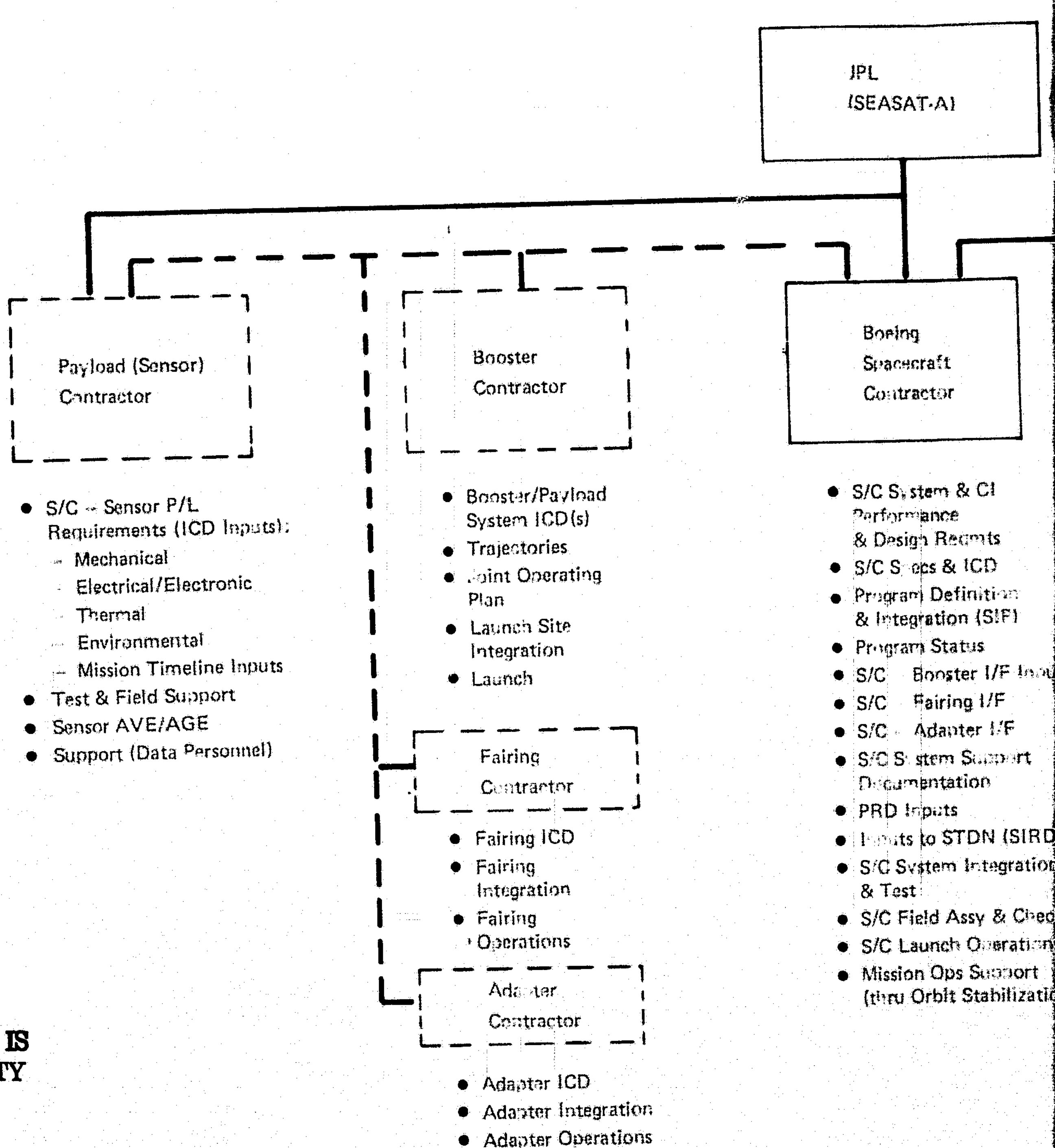

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Figure 8.2-2: Spacecraft/SEASAT-A

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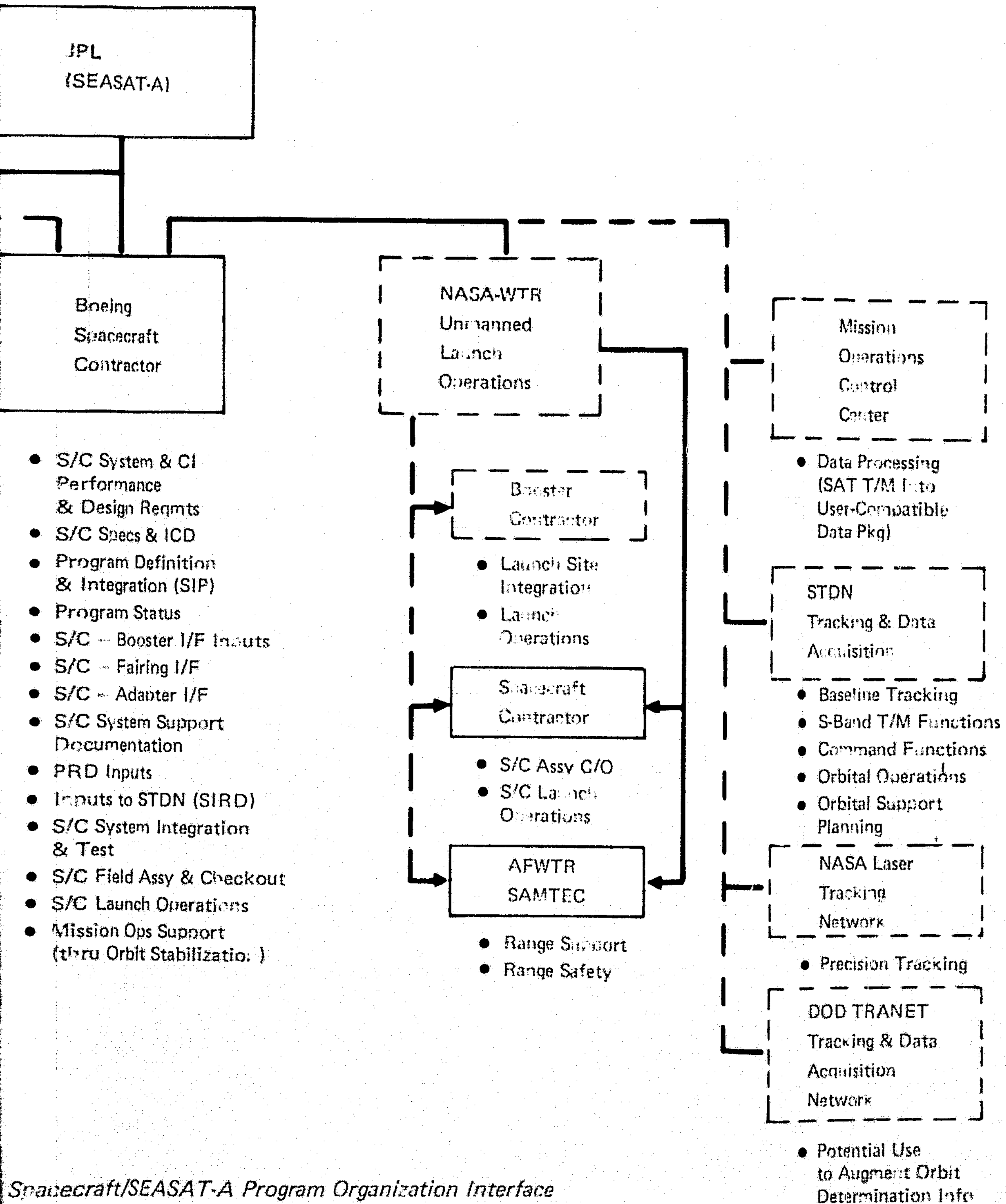

Spacecraft/SEASAT-A Program Organization Interface

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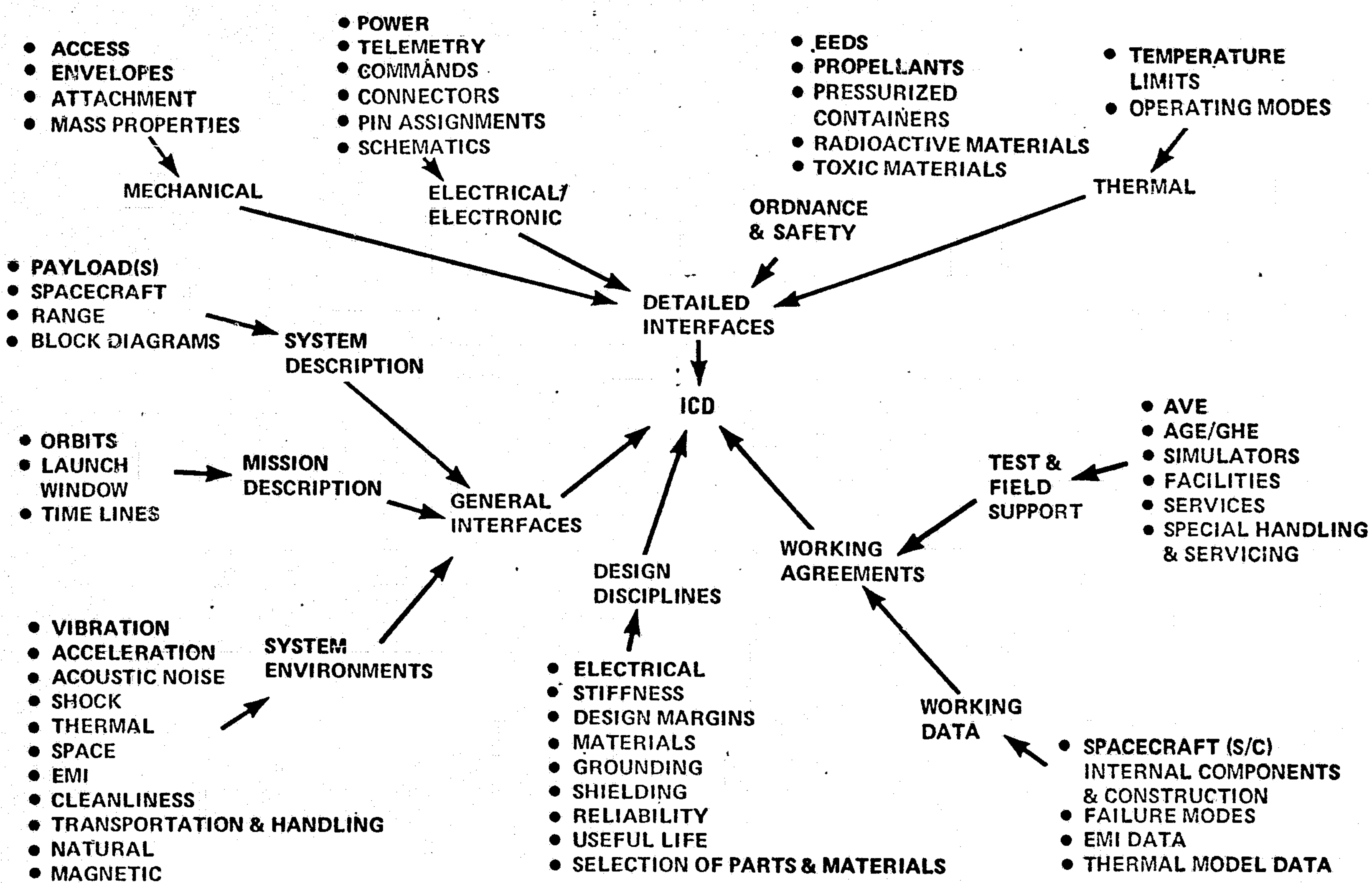

Figure 8.2-3: Interface Control Document

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8.2.2 (Continued)

spacecraft. Physical characteristics will be defined by drawings depicting space allocation, orientation, access limitations, alignments, etc. as well as the physical relationships between the mating surfaces. The functional characteristics, including detail signal characteristics between payloads and the spacecraft telemetry and command subsystems, as shown in Figure 8.2-3, shall be defined in the ICD documents with allowable limits and tolerances for the parameters.

Hardware/software item specifications and layout drawings shall identify the requirements at or across these spacecraft interfaces. The control of interfaces is established through the signed, approved ICD(s).

Typical design disciplines/requirements controlled in the ICD(s)/specifications are as follows:

- o Grounding and bonding in accordance with MIL-B-5087B,
- o Payload(s) to comply with the EMI characteristics of MIL-STD-461A,
- o Electro-explosive devices (EED) to meet AFWTRM 127-1, and 100 watt per square meter sensitivity,
- o A materials, finishes process plan will be developed and imposed on the spacecraft and interfacing hardware, and
- o Power quality (transients spikes, power line ripple, voltage drift).

8.2.3 Payload Integration. The immediate task following program go-ahead is the refinement and detail definition of interface requirements. This effort produces approved payload, spacecraft and booster ICD's, definitization of hardware specifications, configuration item (CI) specifications, and the spacecraft integration and test plans in time to support the preliminary design review (PRD) and the subsequent configuration item design review (CIDR) baseline review. The PDR becomes the basis for detailed coordination of working interfaces at program interface meetings, including technical interchange working group (TIWG), launch test working group(LTWG), and orbital test working group (OTWG). This interchange of technical information/data results in test planning, test procedures, operations planning including operational support documentation. Inputs to the SEASAT-A Program Requirements Document (PRD), request for WTR range support, inputs to the SEASAT-A Support Instrumentation Requirements Document (SIRD) for NASA Net work support (STDN support configuration), and range safety documents are prepared.

8.2.4 Spacecraft Subsystems Integration. To ensure proper physical and functional compatibility, the SEASAT Spacecraft subsystems will be integrated via the system requirements analysis based on results of the sensor requirements analysis (which identifies and defines all the applicable requirements shown in Figure 8.2-3), mission requirements analysis, and design analyses. Final integration will be verified by test. The integration program provides:

8.2.4 (Continued)

- o Allocation of the system level requirements to the satellite subsystems.
- o Definitization of the subsystem requirements and performance parameters.
- o Baseline for subsystems analyses during the definition phase, i.e., configuration item specifications.
- o Maximize use of flight-proven subsystems and equipment.
- o Performance, design, test, and acceptance requirements for SEASAT satellite delivery.

The Contractor will develop, from the systems requirements analysis and the mission and payloads requirements analysis, the detailed system and subsystem airborne vehicle equipment (AVE) and aerospace ground equipment (AGE) performance and design requirements. Product acceptance requirements of the as-built deliverable hardware will be developed and specified in the configuration item product fabrication specifications. A system diagram document will document system level spacecraft diagrams. Schematics will be developed to control system, subsystem and component functional interfaces and will be documented. These schematics will be developed early in the program to ensure interface compatibility, understand the subsystems design, identify design interface problems, trace hardware functions in development of hardware acceptance requirements, and facilitate troubleshooting during satellite systems checkout. Similarly, schematics will be developed and provided for the interconnecting AGE.

8.2.5 Integration Testing. All payload/satellite interfaces will be functionally validated prior to their interconnection. Payload equipment will be operated only in the presence of payload personnel or with their written approval. Test sequences will be jointly agreed upon by Boeing and the payload contractor. Procedures for tests involving operation of the payload will be available for review by the payload contractor at least one week prior to the start of the test. After an interface has been verified by test, each contractor will have the responsibility of maintaining his hardware interface in the as-tested condition.

8.2.6 Mission Operations Interface. A detailed on-orbit mission operations plan will be supported by providing a spacecraft to total mission operations interface. The spacecraft to mission interface will provide data on the following:

8.2.6 (Continued)

- Telemetry subsystem data formatting and calibration for extracting payload and Bus housekeeping data for subsequent display and processing.
- Tracking subsystem characteristics to assure compatibility with NASA and DOD tracking facilities with ephemeris accuracy requirements.
- Command subsystem command formats and capabilities for operation of payloads and Bus Module in accordance with on-orbit operations plans.
- "Red line" limits on Bus subsystems to establish flag alarms.
- Contingency operations capabilities of the Bus to support development of contingency plans in the event "Red line" limits are flagged.
- Attitude Determine and Control (AC&D) software interface compatibility with the overall mission data processing software.

In addition to the above, all data tapes from Bus, payload and spacecraft integration, environmental and acceptance testing will be made available to NASA for support of mission Operations development testing. Detailed interfaces, training and testing of the AC&D software will be provided to NASA operations teams prior to launch as well as technical support from launch thru initial acquisition of nominal attitude on-orbit, estimated as taking up to one-week.

8.2.7 Booster Interface and Integration. Technical integration between elements of the booster system, fairing system, and spacecraft system is shown in Figure 8.2-6. The process is typically illustrated for the existing, established Atlas F Booster/Spacecraft interface, but is readily adapted to the contract baselined booster. For the proposed SEASAT baseline, the satellite system interface is with the Atlas F and the interface plane is at Atlas Station 468. These interfaces for SEASAT boost configuration using the Atlas are the same as the flight-proven USAF P72-1. Hence, the electrical and mechanical interfaces are as defined in the existing USAF System Performance and Design Requirements specification SS-223655A and General Dynamics Report No. GDCA-BGJ71-006.

For the satellite system interface with the Delta booster, proposed alternate, the interface is at Delta Station 604.2. This attachment would be a Marmon clamp of the spacecraft system to the 3731A Universal Attach Fixture. The electrical and mechanical interfaces would be considerably different. Electrically, the interface would be programmed and powered by the booster to the separation clamp. Mechanically, the joint would fit the existing 3731 Attach Fitting.

Launch site integration is typically depicted in Figure 8.2-7. The mutual support and division of responsibilities, organization control, data flow, and schedules will be defined in the SEASAT System Integration Plan.

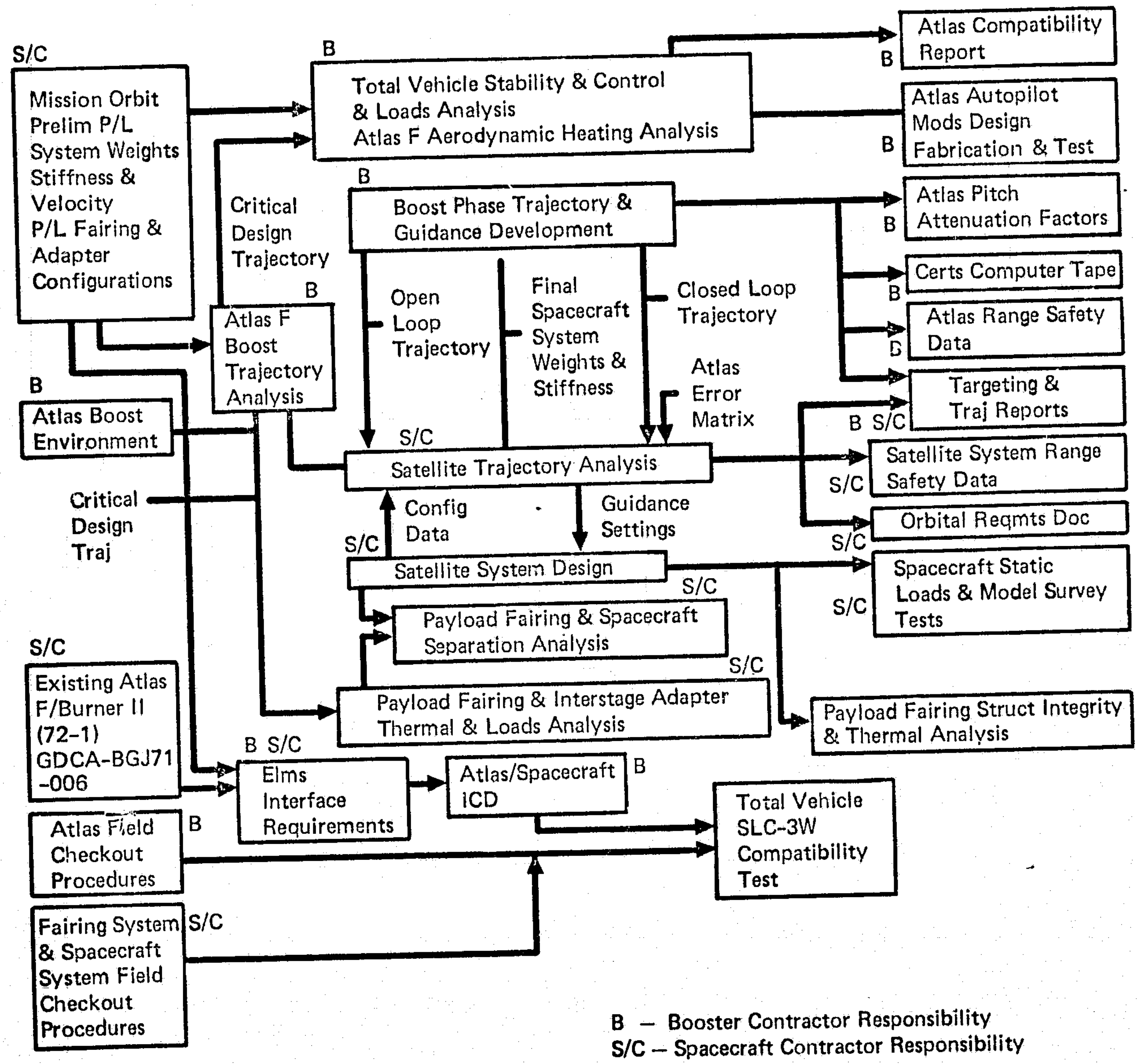

Figure 8.2-6: Atlas F/Satellite System Integration is Proven

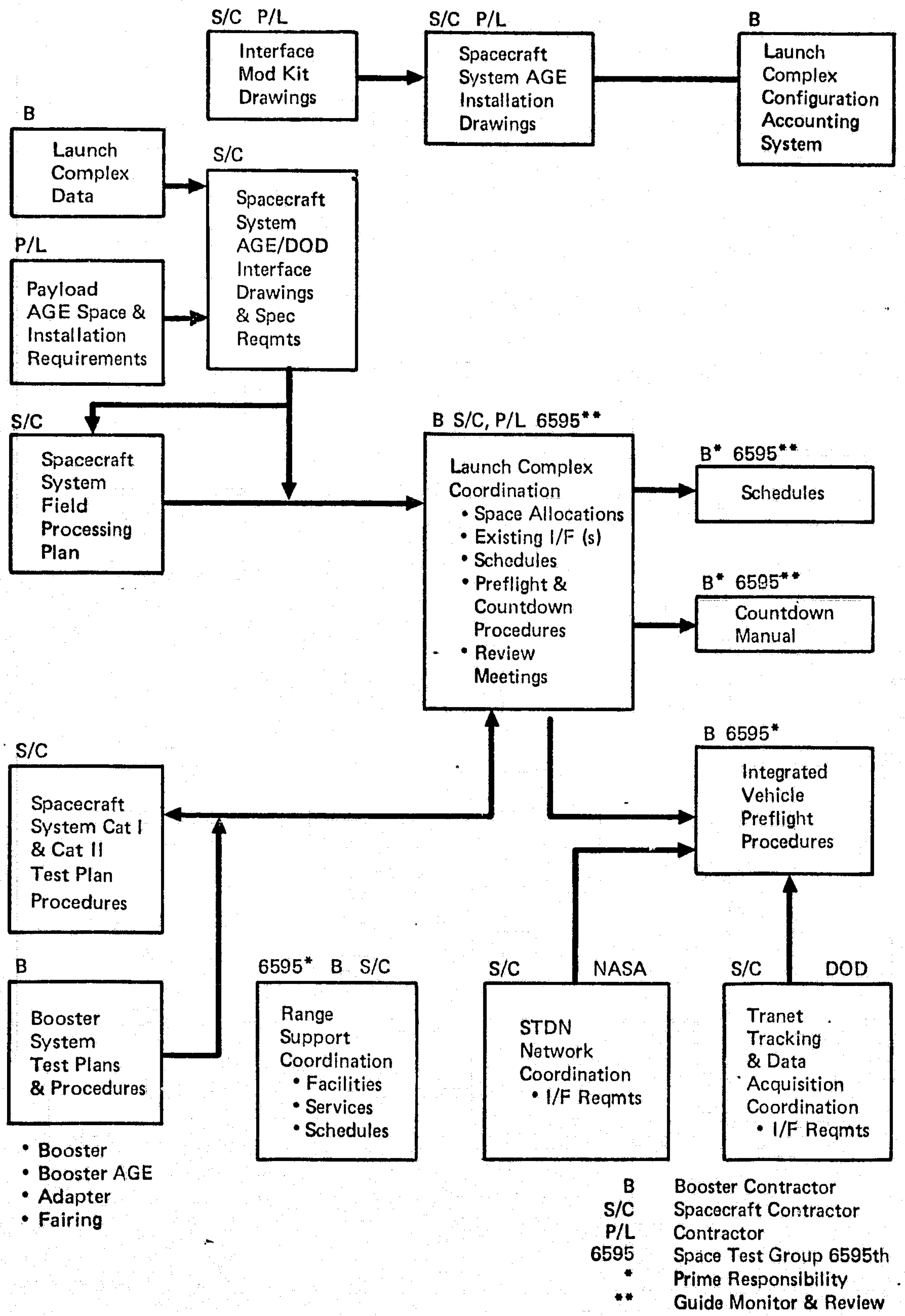

Figure 8.2-7: Launch Site Integration is Proven

8.3 QUALITY ASSURANCE

The SEASAT Quality Program will incorporate the existing Boeing Quality Control system defined in D2-4800, the Boeing Quality Control Manual. This system has been proven on a number of spacecraft programs including Burner II, Burner IIA, SESP 72-1, S-3 and MVM 73-1.

Through implementation of the existing Boeing Quality System, as modified by the SEASAT Quality Plan, the following baseline milestones are recognized as fundamental to a quality spacecraft system.

- Development and release of quality designs.
- Selection, approval, and control of experienced suppliers.
- Verification of specifications and drawings.
- Establishment and maintenance of configuration control.
- Detailing of inspections, tests, etc., in planning orders.
- Component and system qualification.
- Delivery to customer.
- Integration test support.
- Checkout support to enable spacecraft release for flight.

As program schedules and requirements are established, these and other milestones will provide a basis for program quality management.

The Boeing Quality Control program is characterized by separate, but closely interrelated, organizational activities reflecting Engineering responsibility for design quality, Manufacturing and Materiel responsibility for providing quality hardware, and Quality Control for ensuring conformance of the spacecraft, payload integration and associated support equipment to Quality standards.

The Quality Control organization for this program is structured to be responsive to program needs, thus providing immediate identification and evaluation of quality problems and their resolutions. It will be established around a program-oriented quality control group comprised of a nucleus of experienced personnel having spacecraft experience in fabrication, assembly, test, and launch operations.

PROGRAM MANAGEMENT

QUALITY CONTROL MANAGER

Quality Control Engineers
Quality Control Planners

AEROSPACE GROUP DIRECTOR OF QUALITY CONTROL

All Program Q.C. Managers
Quality Control Manual
Budgets and Estimates
Quality Data Group
Statistical Quality Control
Metrology & Quality Control Labs
Source Quality Control
Production Inspection

8.3 QUALITY ASSURANCE (CONTINUED)

As depicted above, the SEASAT Quality Manager has the total resources of the Quality Control Director to draw on for support, thus providing for Program economy without sacrificing program dedication and awareness.

Consistent with the organizational structure, maximum emphasis will be placed on direct personal contact between planners, designers, materiel buyers, manufacturing engineers, and liaison engineers. These personnel are collocated, and the vast majority of interfacing is person to person.

The SEASAT Quality Manager will direct quality control activities and monitor the implementation and conduct of the quality plan. He will review and analyze quality cost data and maintain a cost-effective quality control program. Quality Control manpower support will be agreed to and requested by each technical subsystem manager who has the budget control for all effort associated with his subsystem. He will analyze discrepancy data and ensure that adverse trends are detected and used to adjust the conduct of fabrication, assembly, and test inspection operations.

8.4 RELIABILITY PROGRAM PLAN

The reliability of "one mission only" vehicles can be assured only by the rigorous application of proven procedures and controls for each step in the total program. The reliability program proposed for SEASAT is very similar to those approved by the Air Force and successfully implemented on the STP P72-1 and Small Satellites (S3) programs, and will provide this step by step control during the design, manufacturing and test phases of the program.

The SEASAT Reliability Program will be based on the existing S3 Reliability Program Plan, D233-10006-1, revised as necessary to satisfy specific SEASAT-peculiar requirements (meetings, data submittals, etc). This approach will assure early implementation of the program tasks, consisting of management controls, analyses, parts controls and other design reliability assurance activities.

Reliability Program Management Organization. The SEASAT System Assurance - Reliability Organization will have the responsibility for the reliability program. The working relationship of the SEASAT reliability organization with other organizations will be similar to that existing for the S3 program, and detailed in the S3 Reliability Program Plan. The Reliability Staff, Aerospace Research and Engineering Division, will provide technical assistance, as required, including reliability specialists services and reliability data.

Supplier and Subcontractor Reliability Program. The existing S3 Reliability Plan defines responsibilities for source selection, supplier and subcontractors reliability programs and their surveillance, and failure reporting and corrective action by suppliers.

The reliability requirements for suppliers and subcontractors will be defined in the procurement specifications and control drawings; these will establish requirements for preparation or updating of the Reliability Program Plans of major subcontractors.

Reliability Analysis. All reliability analyses will be in accordance with MIL-STD-756A procedures. This entails establishment of a reliability model; reliability requirements allocation to provide quantitative guidelines; reliability design prediction to provide detail design guidance; reliability assessment to demonstrate compliance with reliability requirements.

Design Review Activities. Design reviews will assess the reliability achieved in the design, and will review results of trade studies, effects of engineering decisions and changes on reliability. Critical processes and potential problem areas will be identified.

The contractor will also conduct in-house design reviews prior to drawing release. Reliability personnel will participate in these reviews and will also continuously review the design during development. The Reliability Organization will prepare reliability requirements for inclusion in the system, and procurement specifications. The drawings and specifications will then be reviewed and approved to assure conformance with the reliability requirements.

8.4 RELIABILITY PROGRAM PLAN (CONTINUED)

Failure Mode and Effect Analysis (FM&EA). A FM&EA will be accomplished by the contractor as a basic approach to attainment of reliability in the design. The requirements for a high probability of success for a 12 month mission indicates the need for this FM&EA. This analysis will assist the designer's effort to eliminate or minimize the cause of potential failures, to optimize duty cycles and backup operating modes, and to verify redundancy effectiveness. A system level preliminary FM&EA will be performed by the reliability organization to further define the depth and requirements for additional analysis. The FM&EA will include the satellite system during on-orbit operation and will be used to determine back-up command and operating modes.

Parts Selection and Control. Boeing maintains a high reliability preferred parts list for general project application (Section 500 of the company's Parts Standard Document, D590). High reliability parts from this preferred parts list will be used as a basic step towards achieving the required high reliability for the mission.

A "Parts Selection" procedure will be used for the SEASAT Program to identify existing Boeing preferred parts and provide a supplementary list of TX and ER parts needed for SEASAT requirements. The procedure will also provide for equivalent requirements for subcontractor parts selection.

The selection of new parts, including sugcontractor design, requires the approval of the Reliability Assurance Organization which supports project design with recommended derating, reliability evaluation and the review of electronic parts stress analysis including application criteria. This review function by a reliability parts application specialist will assure the correct part selection for meeting reliability requirements.

Defective/ Suspect Parts and Materials. All suspect material notices received from the Air Force and in-house sources during the contract period will be reviewed for applicability to SEASAT design and for possible corrective action, if required, as defined in the Reliability Program Plan.

Failure Reporting, Analysis and Corrective Action. The contractor's existing failure reporting, analysis and corrective action system, defined in the S3 Reliability Program Plan, will be utilized. The subcontractor failure reporting and corrective action responsibilities will parallel and complement the Boeing in-house procedures.

Positive corrective action will be assured by the assignment of responsibility based on the analysis of the failure conditions, probable causes, and the most appropriate organizations to accomplish corrections, including the contractor or supplier.

The status of the failure reports, corrective action and close-out are subject to review at scheduled management meetings, at the pre-acceptance review and at the Flight Readiness Review; these reviews are intended to verify close-out of all reports, and to support flight readiness evaluation.

8.4 RELIABILITY PROGRAM PLAN (CONTINUED)

Reliability Testing and Demonstration. Tests performed solely for the purpose of demonstrating reliability requirements or obtaining reliability MTBF data will not be accomplished. However, all testing is structured to provide the data required to assure conformance with performance and environmental requirements of the specification. The Reliability Organization will review and approve test plans and procedures to ensure the adequacy of the design and equipment. Test results will be evaluated to identify problem areas and effects of failures on the system.

8.5 PROCUREMENT PLAN

The Make or Buy Program, Subcontract Management Plan, and Parts Procurement Activity are described in this section. Make or Buy planning identifies the work to be performed by Boeing (Make) or by outside suppliers (Buy) allowing Boeing Aerospace Company to achieve its business objectives. The Subcontract Management Plan presents our proposed control of the subcontractors. The Parts Procurement Activity defines the procurement, acceptance, and release of parts and raw material. Boeing's procurement procedures are fully compatible with all applicable laws and regulations.

8.5.1 Make or Buy Program. The requirement for procurement processing originates from actions of the Make or Buy Committee, and results in the placement of subcontracts or lesser value categories of purchase orders. The Make or Buy Committee for the SEASAT Program is established by the Program Manager and represents Project Engineering, Purchasing, Manufacturing, Quality Control, Facilities, and Business Management. All Boeing Make or Buy activities are conducted in accordance with the provisions of the Boeing Aerospace Company Make or Buy Guide D2-121359-1.

The following criteria is considered in determining the Make or Buy structure:

- o Design sensitivity of the item (firmness of specification and potential impact on other items).
- o Industry costs versus in-house costs.
- o Industry capability versus in-house capability.
- o Existence of qualified equipment from previous programs.
- o Benefit to customer through employment of specially qualified sources.

Prime consideration for the SEASAT Program was directed to suppliers with qualified equipment and make items directed to areas where changes could most readily be incorporated with least program perturbation.

8.5.2 Subcontract Management Plan. In order to provide a meaningful basis for selecting potential subcontractors, procurement packages are prepared and sent to previously screened bidders with detailed instructions for preparation of technical, management, and cost proposals, as appropriate. When proposals are received, sections such as engineering, management, and finance are distributed to appropriate SEASAT Program areas for evaluation. The evaluations are returned to the Procurement Manager who, in cooperation with other functions, uses them to select the preferred subcontractor. In programs of this type, it is Boeing's policy to enter into firm fixed price subcontracts to the maximum extent possible. The negotiated subcontract package will consist of four elements to facilitate monitoring, control, and flow down of customer requirements, they are:

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8.5.2 (Continued)

- Technical Definition
- Supplier Data Requirement List
- Delivery Schedules and Quantities
- Legal and Administrative

Monitoring and control is accomplished by the subsystem team, headed by the technical subsystem manager, who coordinate in directing the activities of the subcontractor and in resolving any problems that occur. Their primary task is to provide a real time interface between the subcontractor and the program and verify that all contractual requirements are met in a timely and cost effective manner. In addition to the subsystem manager's responsible, the team includes the buyer and assigned specialists from Quality Control, Finance and support functions of Materiel. Specific tasks to be accomplished include:

- Assist the subcontractor in the interpretation of Boeing requirements as set forth in contractual documents and specifications.
- Assist the subcontractor in identifying and obtaining the required inputs from Boeing, i.e. data approvals, test procedure approvals, test result approvals, government furnished equipment, etc.
- Review and/or approve all critical lower tier procurements placed by the subcontractor.
- Review and monitor data, information, and milestones within their respective areas submitted in accordance with Data Requirements Lists to ensure that the subcontractor is performing adequately to satisfy contract objectives.
- Maintain an effective and agressive follow-up communication by telephone and visit with respective subcontract counterparts pointing out areas of potential problems affecting quality product design, costs, schedules, or any other factors relating to the contract.
- Monitor and report weekly on subcontractor progress to the technical subsystem manager and program manager. Provide early identification of problems and assist in resolution.
- Assist the subcontractor in preparing for program reviews, design reviews, and technical interchanges, ensuring that all contract data requirements are met.
- Monitor the subcontractor's approved Quality Assurance program for compliance.
- Conduct selected mandatory inspections on in-process hardware.
- Monitor configuration management practices and ensure compliance with subcontract requirements.

8.5.2 (Continued)

- o Review and approve of subcontractor test procedures. Provide approval of restarts found necessary during test, thus eliminating cost and schedule slides inherent in a normal approval cycle.
- o Witness developmental, qualification, and acceptance testing from both quality and technical viewpoints. Review and accept or reject test data, and participate in the review and acceptance of deliverable hardware.

8.5.3 Parts and Raw Material Procurement Activity. All purchased parts and raw materials required for the program will be purchased by the Central Procurement organization of the Boeing Aerospace Company. All parts and raw materials called out on Engineering Drawing Parts List are released for procurement through a computer controlled system by the Manufacturing Planning Group. These requirements are first screened for available inventory and then forwarded to the specific buying group for procurement. A weekly status listing of all potential and actual shortages to Manufacturing need dates is published to ensure continued effort by Engineering, Purchasing and Shortage Controllers to resolve all program shortages. All purchased equipment, parts and raw materials are inspected and screened by Boeing Aerospace Company Quality Control upon receipt according to their classification, e.g. Hi-Rel, vendor part number, etc.

The following is a preliminary Make or Buy structure covering "Important Items".

MAKE

Solar Panel (Substrate)

Command Relay Box

TT&C Antenna

Bus Module Structure

Payload Module Structure

AGE

BUY

Solar Panels (Cells)

Power Conditioning

Solar Array Drives

Scan Wheel

Attitude Control Electronics

Stick Array Antenna

Radio Frequency Subsystem

Processor

Tape Recorder

Command Unit

9.0 LAUNCH VEHICLE

The proposed launch vehicle consists of an Atlas F, a flight proven adapter system and a standard Burner II (STP P72-1) upper stage. The Burner II with the SEASAT spacecraft mounted on top are enclosed in a Burner II (STP P72-1) heat shield with a 145 inch long cylindrical section - see Figure 3.1-1.

All launch vehicle integration AGE and GHE are available both in Seattle and at VAFB (Western Test Range). Also the Atlas F pads at VAFB currently accept the Burner II stage and fairing interfaces. The launch vehicle system has high design margins and provides a high launch availability dependent upon the season of launch.

This launch vehcile configuration has been flown successfully in the past and offers the advantages of existing interfaces, existing test procedures and proven reliability. This configuration also allows verification of physical and electrical interfaces between the spacecraft, Burner II and the heat shield prior to shipment to VAFB.

The Atlas F/Burner II launch vehicle was chosen because it is believed to represent a comparable, or lower, cost launch vehicle than Delta and requires no interface development between the spacecraft and launch vehicle. However, the proposed spacecraft can be readily designed to interface with Delta and, should the imaging radar become a firm payload, selection of Delta appears mandatory to provide the shroud volume required to accommodate the imaging radar.

10.0 IMAGING RADAR INCORPORATION

Should it become necessary to incorporate the Imaging Radar sensor in the SEASAT-A, the impact to the configuration and the program cost is considered to be sufficiently significant to jeopardize the "low cost" SEASAT-A. The radar subsystem considered herein is what appears to be the preferred configuration (Option 2) as covered in JPL letter 672-LDA:vr, dated 1 March 1974. The principle and obvious areas of impact are:

Antennas:

- o Design and Deployment
- o Installation on Payload Module
- o Deployment for Unobstructed Viewing by all Antennas
- o Solar Panel Shadowing

Data Rate:

- o Video Tape Recorder (e.g.; ERTS Type)
- o Dedicated Wideband Data Link Required

D. C. Power:

- o Power Subsystem 100% Larger

Test:

- o Component/Subsystem Level
- o Subsystem Level
- o Wideband Data Link SE

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10.1 SPACECRAFT CONFIGURATION

The spacecraft configuration incorporating Imaging Radar is shown in Figure 10.1-1. The configuration is generally similar to the proposed spacecraft (Figure 3.1-1) except as follows:

(a) Two (2) 4 X 2.5 meter phased arrays are included. These arrays are folded in half along the 4 meter axis and stowed in a forward/aft position underneath the (5) stick antennas.

(b) The "Bus Module" will probably require an increase in volume to accommodate the additional power subsystem components since the addition of Imaging Radar requires a substantial increase in power.

(c) Addition of the two (2) 4 X 2.5 meter phased arrays prohibits use of the proposed launch vehicle (Atlas F, Burner II and Burner II heat shield). Consequently, the Imaging Radar configuration is shown on a Delta launch vehicle with the 96.0" O. D./86.0 I.D. Delta fairing.

The "On Orbit" configuration with phased arrays and stick antennas deployed is shown in Figure 10.1-2. A major problem with this configuration is "shadowing" of the solar array by the (2) 4 X 2.5 meter phased arrays. This "shadowing" problem has not been worked to date.

10.2 TELECOMMUNICATIONS

The additional data handling requirements imposed by the Imaging Radar is not compatible with the proposed TT&C Subsystem and will require the addition of a wideband data handling and communication link. Because of the dedicated requirement for the wideband system, it should be considered a part of, and installed in, the Payload Module. Possible candidate equipment for this system would be the wideband communication system that was used on the ERTS Satellite which includes a wideband video tape recorder.

The ERTS Wideband Telemetry Link for the 15 Megabit Multispectral Scanner (MSS) will be adequate for the Imaging Radar 16 Megabit data. It provides 10^{-6} BER data at about +4 dB signal margin using a 10 watt transmitter. The orbit altitudes for both ERTS and SEASAT are very nearly the same hence the ERTS Wide Band Telemetry subsystem, with minor SEASAT peculiar modifications, could be a good choice for SEASAT. Figure 10.2-1 shows a simplified block diagram of the suggested system.

Digital data ($\approx$16 mbps) from the Imaging Radar is supplied to the wideband tape recorder for data storage and subsequent playback to the ground station. The recorder is a transverse scan, rotary head video recorder with two auxiliary longitudinal tracks available. A data switch can select either realtime data or stored data and provides premodulation filtering of the signal to be transmitted.

The 3 x 10^{10} Bit storage capacity of the ERTS Video Recorder exceeds the SEASAT requirement of about 5 x 10^9 Bits. The supplemental JPL letter announcing additional Imaging Radar requirements does not exactly agree with JPL presentation viewgraphs dated 27 January 1974.

Table 10.2-1 is Boeing's understanding of the three data modes based upon the JPL viewgraph calculations and letter requirements. Transmission rates shown in Table-10.2-1 are based upon a maximum 16 Megabit data rate permissible in either Real-Time (R-T) or Playback (PB) modes constrained by a ten minute Playback dump speed-up of the recorder. This allows a full storage period per orbit to be dumped over one ground station during a ten minute pass. A more definitive configuration can be made as the requirements become more clearly defined.

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TABLE 10.2-1

IMAGING RADAR DATA SUMMARY

PARAMETER	MODE I	MODE 2	MODE 3
SWATH WIDTH	100 KM	200-300 KM	10 KM
RESOLUTION	25M	100M	25M
DIGITALIZATION	4 BIT	1 BIT	1 BIT
STORAGE TIME/ORBIT(MAX)	5 MIN.	80 MIN.	40 MIN.
BIT RATE	16 MEGABIT	675 KBPS	800 KBPS
BIT STORAGE REQUIRED (APPROX.)	5×10^9	3.5×10^9	2×10^9
TRANSMISSION RATE MULTIPLICATION			
R-T	1:1	1:1	1:1
PB	1:1	8:1 to 24:1	4:1 to 20:1

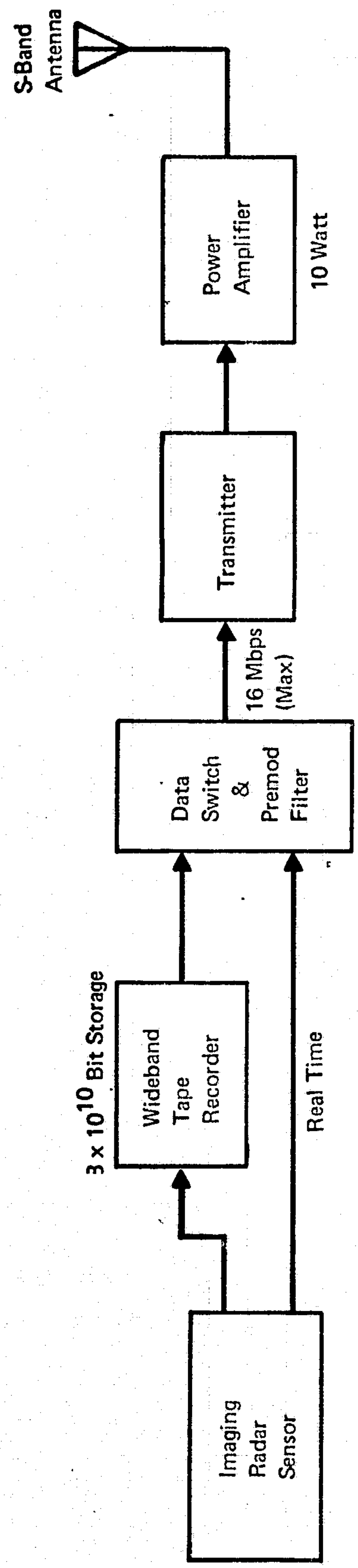

Figure 10.2-1: Wideband Data Handling System

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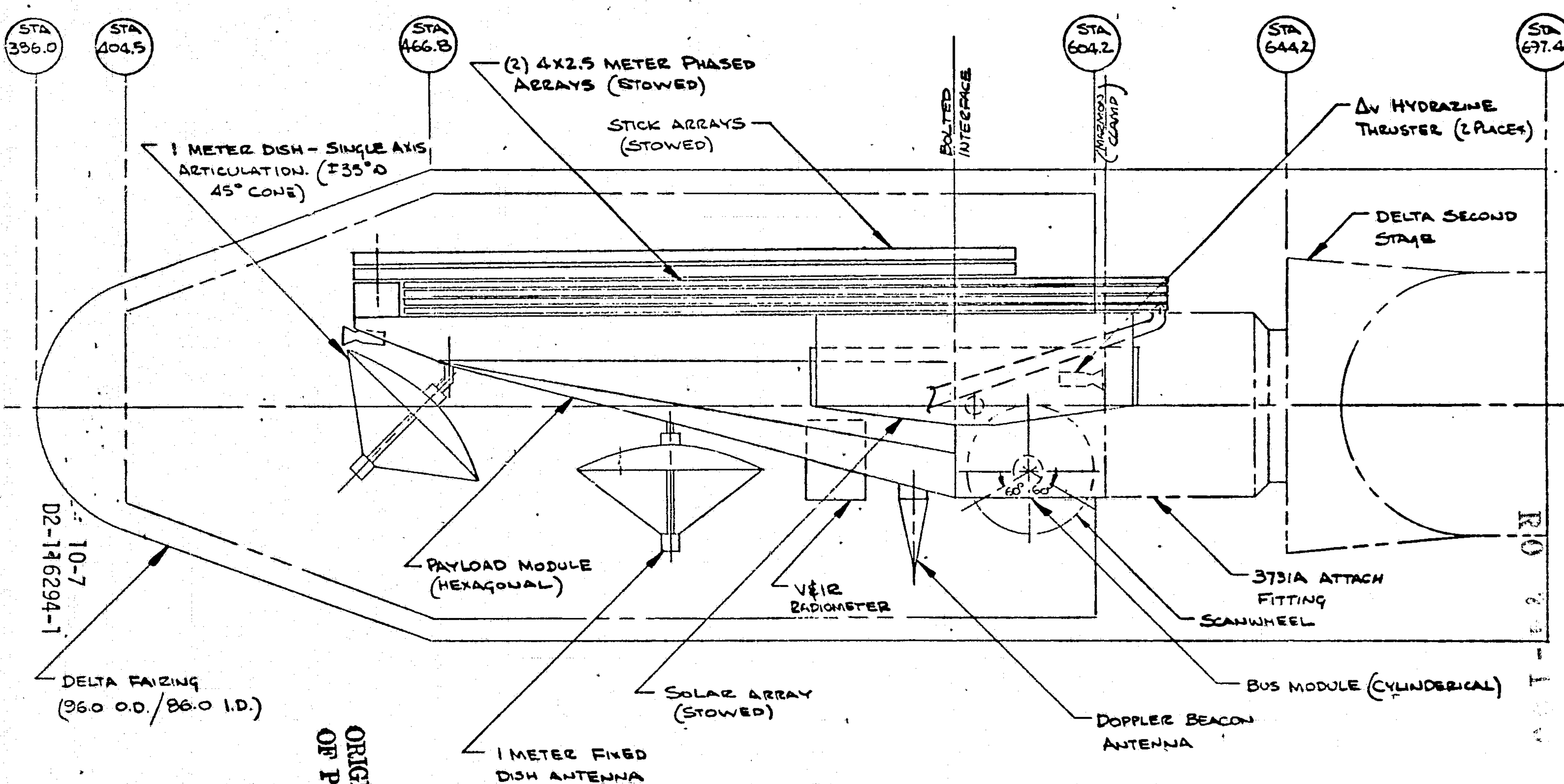
10.3 PAYLOAD ANTENNAS

Option 2 from Attachment 1 of JPL Letter 627-LDA:vr dated 1 March 1974 was selected as it appears to provide a more optimum system for the User and will fit (folded) into the Delta size shroud. To provide unobstructed viewing for all antennas it may be necessary to deploy to the aft of the S/C Bus as shown in Figure 10.1-

It was not possible, in the time available, to conduct a preliminary design of the two phased arrays. However, it is suspected that considerable difficulty will be encountered in the design development to accomplish the phasing necessary for the three swath widths (25Km, 100Km, and 200-300Km). Other than the design development activities, the imaging radar antenna impact to SEASAT-A appears to be associated with the S/C configuration, as discussed in 10.1 above.

10.4 POWER SUBSYSTEM

Addition of imaging radar subsystem to the space vehicle increases the required electrical power by 300 watts. This additional power drain will require an additional 33 sq. ft. of solar array, increase in battery capacity to 80 ampere hours and corresponding sizing changes to the boost and shunt regulators and battery chargers. This increase in size will impact subsystem weight fractions and thermal design on nearly a linear basis.



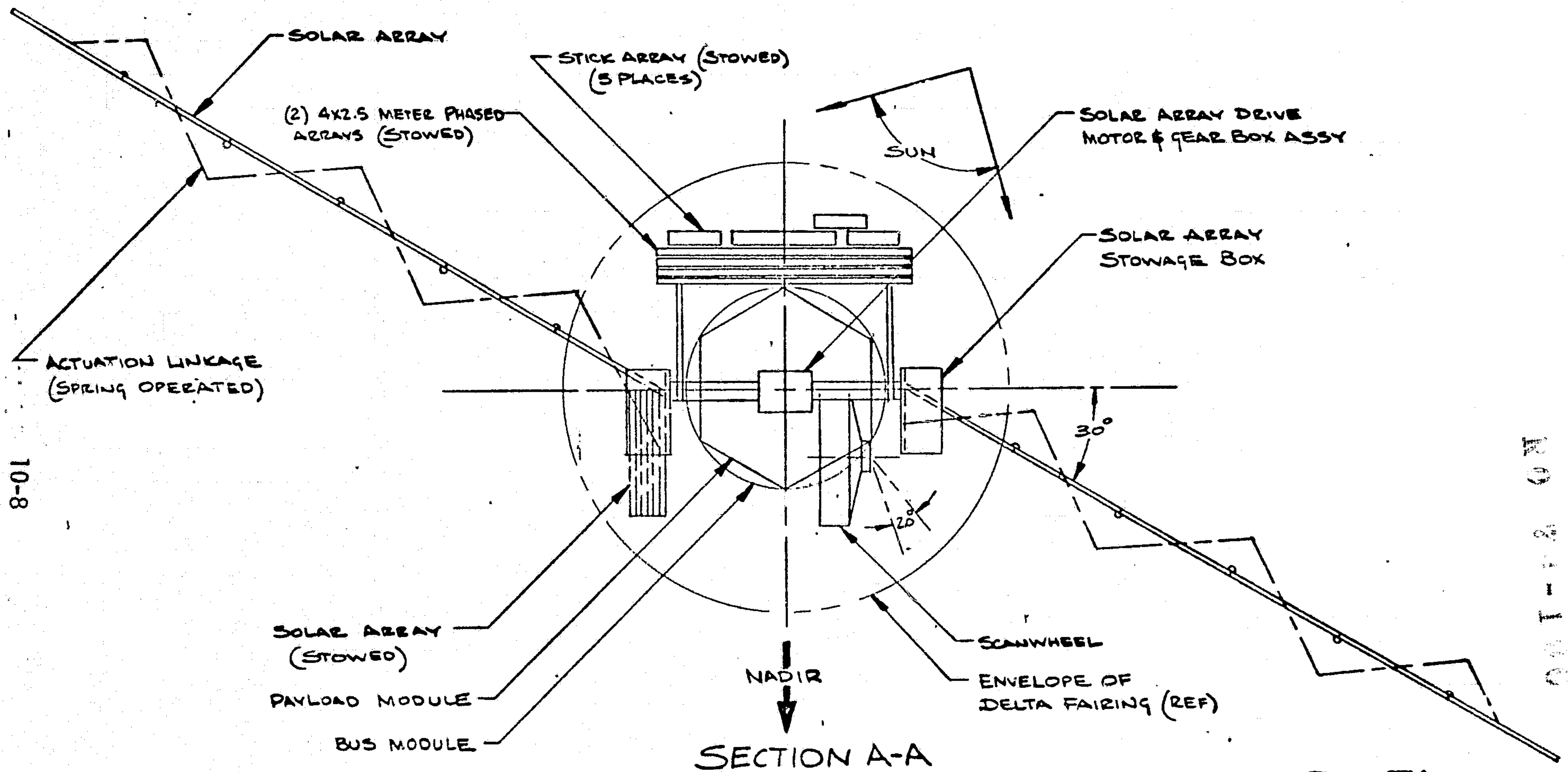
— SIDE VIEW —

— SEASAT, DELTA CONFIGURATION WITH IMAGING RADAR —

SCALE 1:20

FIGURE 10.1-1 SH 1 OF 2

a.Wat 3/5

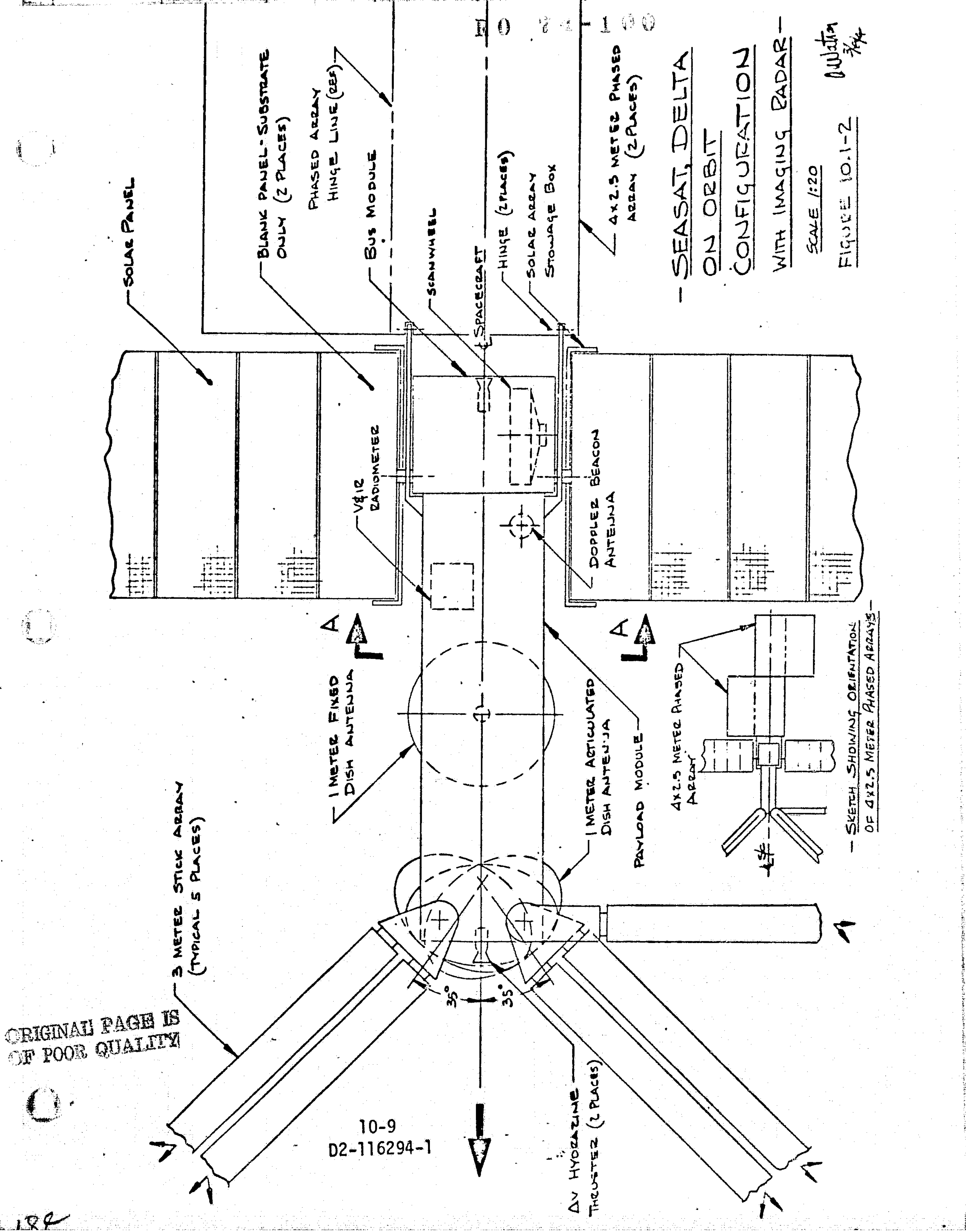


SECTION A-A

SEASAT, DELTA
CONFIGURATION
WITH IMAGING RADAR

SCALE 1:20

FIGURE 10-1-1 SH 2 OF 2



SEASAT, DELTA
ON ORBIT
CONFIGURATION
WITH IMAGING RADAR
SCALE 1:20
FIGURE 10.1-2
SOLAR PANEL
BLANK PANEL-SUBSTRATE ONLY (2 PLACES)
PHASED ARRAY HINGE LINE (REF)
BUS MODULE
SCANWHEEL
℄ SPACECRAFT
HINGE (2 PLACES)
SOLAR ARRAY STOWAGE BOX
4x2.5 METER PHASED ARRAY (2 PLACES)
V&12 RADIOMETER
DOPPLER BEACON ANTENNA
1 METER FIXED DISH ANTENNA
1 METER ARTICULATED DISH ANTENNA
PAYLOAD MODULE
3 METER STICK ARRAY (TYPICAL 5 PLACES)
ΔV HYDRAZINE THRUSTER (2 PLACES)
35°
35°
A
A
SKETCH SHOWING ORIENTATION OF 4x2.5 METER PHASED ARRAYS
4x2.5 METER PHASED ARRAY